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PROCEEDINGS OF THE 1997 USAF AIRCRAFT STRUCTURAL INTEGRITY PROGRAM CONFERENCE

VOLUME I

ASIP

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USAF Aircraft Structural Integrity Program Conference
Hyatt Regency San Antonio
San Antonio, Texas

AUGUST 1998

FINAL REPORT FOR PERIOD 2-4 DECEMBER 1997

MATERIALS AND MANUFACTURING DIRECTORATE
AIR FORCE RESEARCH LABORATORY
AIR FORCE MATERIEL COMMAND
WRIGHT-PATTERSON AFB OH 45433-7734

NOTICE

This technical report has been reviewed and is approved for publication.

Gary K. Waggoner

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Chief
Systems Support Division

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13. ABSTRACT *(Maximum 200 words)*
This report contains the proceedings of the 1997 USAF Structural Integrity Program Conference held at the Hyatt Regency Hotel in San Antonio, Texas, from 2-4 December 1997. The conference, which was sponsored by the Aeronautical Systems Center's Engineering Directorate and the Air Force Research Laboratory's Air Vehicles and Materials and Manufacturing Directorates, was hosted by the San Antonio Air Logistics Center Aircraft Directorate, Aircraft Structural Integrity Branch (SA-ALC/LADD). This conference, as in previous years, was held to permit experts in the field of structural integrity to communicate with each other and to exchange views on how to improve the structural integrity of military weapon systems. Sessions were primarily focused on analysis and testing, engine structural integrity, structural materials and inspections, structural repair, and force management. This year, as in previous years, our friends from outside the U.S. borders provided the audience with outstanding presentations on activities within their countries. It is anticipated this conference will include their contributions in the agenda of future meetings.

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TABLE OF CONTENTS
Volume I

SESSION IV - BONDED COMPOSITE REPAIRS

SESSION V - NONDESTRUCTIVE EVALUATION/INSPECTION

SESSION VI - FATIGUE AND CRACK GROWTH

LUNCH PRESENTATION

VOLUME II

FOREWORD

This report was compiled by the Systems Support Division, Materials and Manufacturing Directorate, Air Force Research Laboratory, Wright-Patterson Air Force Base, Ohio. It was initiated under Task 4349TECA "Corrosion Control & Failure Analysis" with Gary K. Waggoner as the Project Engineer.

This technical report was submitted by the editors.

The purpose of this 1997 Conference was to bring together technical personnel in DoD and the aerospace industry who are involved in the various technologies required to ensure the structural integrity of aircraft gas turbine engines, airframes and other mechanical systems. It provided a forum to exchange ideas and share new information relating to the critical aspects of durability and damage tolerance technology for aircraft systems. The conference was sponsored by the Air Force Materiel Command (AFMC), Aeronautical Systems Center, Deputy for Engineering and Materials and Manufacturing and Air Vehicles Directorates of the Air Force Research Laboratory, Wright-Patterson Air Force Base, Ohio. It was hosted and co-sponsored by the Aircraft Structural Integrity Branch, Aircraft Directorate of AFMC's San Antonio Air Logistics Center, Kelly Air Force Base, Texas.

1997 USAF AIRCRAFT STRUCTURAL INTEGRITY PROGRAM CONFERENCE

AGENDA

MONDAY, 1 DECEMBER 1997

5:00 PM - 7:00 PM	PRE-REGISTRATION

TUESDAY, 2 DECEMBER 1997

7:30 AM - 5:30 PM	REGISTRATION
7:30 AM - 8:15 AM	CONTINENTAL BREAKFAST
7:50 AM - 8:00 AM	OPENING COMMENTS *J. Lincoln*, Aeronautical Sys Ctr

SESSION I - OVERVIEWS
Chairman - *J. Rudd*
Air Force Research Laboratory

8:00 AM - 8:30 AM	ECONOMIC IMPLICATIONS OF DADTA FOR MILITARY AIRCRAFT OPERATORS OF NON-USAF FLEETS *K. Schrader, O. Burnside and W. Sparks*, Southwest Research Institute
8:30 AM - 9:00 AM	NEW APPROACH FOR FATIGUE LIFE MONITORING RNLAF F-16 FLEET *D. Spiekhout*, National Aerospace Laboratory NLR, The Netherlands
9:00 AM - 9:30 AM	XF-2 (JAPANESE NEXT GENERATION SUPPORT FIGHTER) FULL SCALE STATIC TESTS AND DURABILITY TEST - INTERIM TEST RESULTS *M. Kageyama*, Japan Defense Agency
9:30 AM - 10:00 AM	USING EMERGING COMPUTER HARDWARE, SOFTWARE AND COMMUNICATIONS TECHNOLOGIES IN FLEET MANAGEMENT *R. Giese and G. Herring*, Ogden Air Logistics Center
10:00 AM - 10:30 AM	REFRESHMENT BREAK

SESSION II - LIFE ENHANCEMENT
Chairman - *K. Leikach*
NAVAIR

10:30 AM - 11:00 AM	TERMINATING REPAIR OR RESIZING OF DAMAGED/ DISCREPANT HOLES USING EXPANDED BUSHINGS *L. Reid and J. Restis*, Fatigue Technology Inc.
11:00 AM - 11:30 AM	LASER SHOT PEENING OF METALS: TECHNIQUES AND LASER TECHNOLOGY *L. Hackel and C. Dane*, Lawrence Livermore National Laboratory *J. Daly and J. Harrison,* Metal Improvement Co.

TUESDAY, 2 DECEMBER 1997 (Cont'd)

11:30 AM - 12:00 PM DEVELOPMENT OF LASER SHOCK PEENING OF AIRFOILS LEADING EDGE FOR SINGLE ENGINE WEAPON SYSTEMS
S. Mannava, W. Cowie, T. Compton, GE Aircraft Engines
General Electric Company

12:00 PM - 1:30 PM LUNCH AND PRESENTATION OF THE LINCOLN MEDAL
The National Research Council Report on Aging Aircraft
C. Tiffany, Consultant

SESSION III - BONDED COMPOSITE REPAIRS

Chairman - *W. Elliott*
Warner Robins Air Logistics Center

1:30 PM - 2:00 PM AN EXPANDED ROSE MODEL FOR BONDED REPAIR DESIGN AND ANALYSIS
R. Müller and Major R. Fredel, Center for Aircraft Structural Life Extension, United States Air Force Academy
F. Rose, Aeronautical and Maritime Research Laboratory
Fisherman's Bend, Australia

2:00 PM - 2:30 PM ENVIRONMENTAL EFFECTS ON ADHESIVELY BONDED SYSTEMS
Major L. Butkus and S. Johnson, Georgia Institute of Technology

2:30 PM - 3:00 PM QUALIFICATION OF A BONDED REPAIR TO C-5A FUSELAGE CRACKING UNDER SPECTRUM FATIGUE LOADING
C. Guijt and S. Verhoeven, Center for Aircraft Structural Life Extension, United States Air Force Academy

3:00 PM - 3:30 PM REFRESHMENT BREAK

SESSION IV - BONDED COMPOSITE REPAIRS

Chairman - *Major R. Fredell*
United States Air Force Academy

3:30 PM - 4:00 PM EFFECTS OF BONDLINE DEFECTS AND ENVIRONMENTAL EXPOSURE ON ADHESIVELY BONDED COMPOSITE PATCH REPAIR OF CRACKED ALUMINUM ALLOY SHEET
P. Poole, D. Lock and A. Young
DERA

4:00 PM - 4:30 PM DURABILITY PATCH: HIGH CYCLE FATIGUE DAMAGE DOSIMETER AND VIBRATION RESULTS OF SECONDARY STRUCTURE REPAIR
L. Rogers, J. Maly, I. Searle, R. Ikegami, W. Owen, D. Smith, R. Gordon, and J. Ryan, CSA Engineering Inc.

TUESDAY, 2 DECEMBER 1997 (Cont'd)

4:30 PM - 5:00 PM REPAIR METHODOLOGY DEVELOPMENT FOR THE MH-53E COMPOSITE SPONSON
C. Francis, Naval Aviation Depot, Cherry Point, NC
A. Dobyns, Sikorsky Aircraft

5:00 PM - 5:30 PM RESTORING FAIL-SAFETY IN OLD AIRCRAFT USING BONDED REPAIRS
A. Kerr, Advanced Repair Technology International

6:00 PM - 7:30 PM RECEPTION

WEDNESDAY, 3 DECEMBER 1997

7:30 AM - 8:00 AM REGISTRATION/CONTINENTAL BREAKFAST

SESSION V - NONDESTRUCTIVE EVALUATION/INSPECTION

Chairman - *M. Paulk*
Air Force NDI Program Office

8:00 AM - 8:30 AM QUANTITATIVE ASSESSMENT OF THE DETECTABILITY OF CERAMIC INCLUSIONS IN STRUCTURAL TITANIUM CASTINGS BY X-RADIOGRAPHY
F. Child, D. Phillips and L. Liese, Lockheed Martin Aero
W. Rummel, D & W Enterprises Ltd.

8:30 AM - 9:00 AM ADVANCED ULTRASONIC TECHNIQUES FOR CORROSION DETECTION OF AGING AIRCRAFT STRUCTURES
V. Mustafa, A. Chabaz and D. Robert Hay, Tektrend International Inc.
A. Fahr and D. Forsyth, National Research Council of Canada Ottawa, Canada

9:00 AM - 9:30 AM DEVELOPMENT OF AN ELECTROCHEMICAL SENSOR FOR EARLY DETECTION OF FATIGUE DAMAGE IN AIRCRAFT
S. Hudak Jr., M. Miller and G. Cragnolino, Southwest Research Institute
C. Laird, Y. Li and J. DeLuccia, University of Pennsylvania

9:30 AM - 10:00 AM IN-SITU ELECTROCHEMICAL SENSORS FOR DETECTING CORROSION ON AGING AIRCRAFT
G. Davis, C. Dacres and M. Shook, Dacco Scientific Inc.

10:00 AM - 10:30 AM REFRESHMENT BREAK

WEDNESDAY, 3 DECEMBER 1997 (Cont'd)

SESSION VI - FATIGUE AND CRACK GROWTH

Chairman - *R. Eastin*,
Federal Aviation Administration

10:30 AM - 11:00 AM	PROBABILISTIC STRESS SPECTRUM GENERATION *M. Schleider*, Mercer Engineering Reset Center *R. Jansen*, Warner Robins Air Logistics Center
11:00 AM - 11:30 AM	DAMAGE ACCUMULATION AND SPECTRUM EDITING *E. Tuegel and C. Brooks*, AP/ES Inc.
11:30 AM - 12:00 PM	EFFECT OF COMPRESSION ON CRACK GROWTH RATE IN ALUMINUM *J. Harter*, Air Force Research Laboratory *J. Elsner*, Analytical Services and Materials
12:00 PM - 1:30 PM	LUNCH AND PRESENTATION A Risk Based Approach to the Management of Critical Areas on the IFOSTP FT-55 Full-Scale Fatigue Test Article *Capt J.-F. Leclerc*, Directorate of Technical Airworthiness National Defence Headquarters, Ottawa, Canada *J. Dubuc*, Bombardier Defence Systems Division Mirabel, Quebec, Canada

SESSION VII - FATIGUE AND CRACK GROWTH

Chairman - *J. Gallagher*
Air Force Research Laboratory

1:30 PM - 2:00 PM	DAMAGE TOLERANCE CHARACTERIZATION OF THICK, WROUGHT ALUMINUM PRODUCTS WITH AND WITHOUT STRESS RELIEF; FOCUS ON TOUGHNESS AND CRACK GROWTH PROPERTY MEASUREMENT TO CAPTURE RECENT BREAKTHROUGHS IN FORGING STRESS RELIEF TECHNOLOGY *R. Bucci, R. Bush and G. Kuhlman*, Aluminum Company of America
2:00 PM - 2:30 PM	THE EFFECT OF PRIOR CORROSION DAMAGE ON THE SHORT CRACK GROWTH RATES OF TWO ALUMINUM ALLOYS *A. Taylor*, TexSEM Laboratories *D. Hoeppner*, University of Utah
2:30 PM - 3:00 PM	SHORT AND LONG FATIGUE CRACKS: ANALYSIS AND IMPLICATIONS TO LIFE PREDICTION METHODOLOGY *A. Vasudevan*, Office of Naval Research *K. Sadananda*, Naval Research Laboratory
3:00 PM - 3:30 PM	REFRESHMENT BREAK

WEDNESDAY, 3 DECEMBER 1997 (Cont'd)

SESSION VIII - FATIGUE AND CRACK GROWTH

Chairman - *E. Davidson*
Aeronautical Systems Center

3:30 PM - 4:00 PM	FRETTING AS A FATIGUE CRACK NUCLEATION MECHANISM - A CLOSE-UP VIEW *C. Elliott and D. Hoeppner*, University of Utah
4:00 PM - 4:30 PM	AN EVALUATION OF EMPIRICAL AND ANALYTICAL MODELS FOR PREDICTING FATIGUE CRACK PROPAGATION LOAD INTERACTING EFFECTS *K. Walker*, Aeronautical and Maritime Research Laboratory Melbourne, Australia
4:30 PM - 5:00 PM	A NEW CONCEPT TO DESCRIBE LOAD INTERACTION EFFECTS IN FATIGUE CRACK PROPAGATION *M. Lang*, Air Force Research Laboratory
5:00 PM - 5:30 PM	DETERMINING FLIGHT LOADS AND CRACK GROWTH RATES FROM FAILED AIRCRAFT STRUCTURAL COMPONENTS *D. Shockey, T. Kobayashi and R. Klopp*, SRI International *C. Schmidt*, Hewlett-Packard
5:30 PM - 6:00 PM	FATIGUE TESTING AND NDI OF CORRODED FULL-SCALE FUSELAGE PANELS *D. Jeong, J. Canha and G. Neat*, U.S Department of Transportation *S. Kokkins*, Foster-Miller *T. Flournoy*, Federal Aviation Administration

THURSDAY, 4 DECEMBER 1997

7:30 AM - 8:00 AM	REGISTRATION/CONTINENTAL BREAKFAST

SESSION IX - ANALYTICAL METHODS

Chairman - *C. Harris*
NASA Langley

8:00 AM - 8:30 AM	EVALUATION OF PROGRESSIVE FRACTURE IN WOVEN AND NON-WOVEN COMPOSITE PANELS *L. Minnetyan*, Clarkson University *C. Chamis and P. Gotsis*, NASA Lewis Research Center
8:30 AM - 9:00 AM	NEW GENERATION DESIGN AND ANALYSIS PROCEDURES FOR BONDED REPAIR OF AIRCRAFT *P. O'Donoghue, D. Pipkins and H. Kawai*, Knowledge Systems Inc. *J. Park and S. Atluri*, Georgia Institute of Technology

THURSDAY, 4 DECEMBER 1997 (Cont'd)

9:00 AM - 9:30 AM	ASSESSMENT OF ANALYSIS METHODOLOGIES FOR PREDICTING FATIGUE CRACK GROWTH AND RESIDUAL STRENGTH OF AGING AIRCRAFT *F. Brust and R. Kurth*, Battelle Columbus
9:30 AM - 10:00 AM	CORROSION AND WIDESPREAD FATIGUE DAMAGE OF CRITICAL AIRCRAFT STRUCTURE *D. Tritsch*, University of Dayton Research Institute *Capt D. Groner*, Air Force Research Laboratory
10:00 AM - 10:30 AM	REFRESHMENT BREAK

SESSION X - ANALYTICAL METHODS

Chairman - *F. Bartlett*

U.S. Army

10:30 AM - 11:00 AM	MODELING FASTENED STRUCTURAL CONNECTIONS USING FINITE ELEMENTS *R. Actis and B. Szabo*, Engineering Software Research and Development, Inc.
11:00 AM - 11:30 AM	DEVELOPMENT OF PASSIVE AND SMART C-130 AIRCRAFT STRUCTURAL BOLTS *R. Waldbusser*, Warner Robins Air Logistics Center *L. Thompson*, Strain Monitor Systems, Inc.
11:30 AM - 12:00 PM	PRELIMINARY DESIGN WITH DAMAGE TOLERANCE CONSTRAINTS USING ASTROS *D. Pipkins, P. O'Donoghue and H. Kawai*, Knowledge Systems Inc. *S. Atluri*, Georgia Institute of Technology
12:00 PM - 1:30 PM	LUNCH AND PRESENTATION B-1B Horizontal Stabilizer Substructure Failures *J. Morgan*, Oklahoma City Air Logistics Center

SESSION XI - DYNAMICS

Chairman - *M. Basehore*

William J. Hughes Technical Center, FAA

1:30 PM - 2:00 PM	DEVELOPMENT OF DYNAMIC MODELS FOR THE B-1B HORIZONTAL STABILIZER TO PREDICT RESPONSES FOR ENGINE TAKEOFF NOISE *J. Rosenthal*, Boeing Defense and Space Group
2:00 PM - 2:30 PM	STRUCTURAL FATIGUE OF AIRCRAFT STRUCTURES CAUSED BY RANDOM/ACOUSTIC LOADING SOURCES *N. Bishop*, NCode International
2:30 PM - 3:00 PM	RECENT TECHNOLOGY ENHANCEMENT IN HELICOPTER STRUCTURAL INTEGRITY COMPUTER PROGRAM *C. Crawford and D. Friend*, Georgia Tech Research Institute *T. Christian and G. Chamberlain*, Warner Robins Air Logistics Center

THURSDAY, 4 DECEMBER 1997 (Cont'd)

3:00 PM - 3:30 PM	REFRESHMENT BREAK
	SESSION XII - FORCE MANAGEMENT Chairman - *J. Turner* San Antonio Air Logistics Center
3:30 PM - 4:00 PM	DETECTING HIGH CYCLE FATIGUE WITH USER DEFINED REGIME RECOGNITION *J. Cicero*, Systems and Electronics Inc.
4:00 PM - 4:30 PM	THE ABCs OF NDE DEVELOPMENT AND TRANSITION FOR AGING AIRCRAFT *D. Hagemaier and D. Wilson*, The Boeing Company
4:30 PM - 5:00 PM	C-17A INDIVIDUAL AIRCRAFT TRACKING PROGRAM *R. Seldler and K. Liu*, The Boeing Company
5:00 PM - 5:30 PM	THE ELECTRONIC AFTO 451 *H. Peacock Roland*, Lockheed Martin Aero Systems *TSgt A. Taus*, Altus AFB, OK
5:30 PM - 6:00 PM	FAA MSR/LSR FLIGHT INSPECTION FLEET AIRCRAFT STRUCTURAL INTEGRITY PROGRAM *J. Marks Jr. and J. Abel*, Raytheon E-Systems

ASIP PARTICIPATING EXHIBITORS

Advanced Repair Technology Intl.

Avibank Mfg Inc.

Battelle

Boeing Defense & Space

Boeing MAMS & OO-ALC/LACM

Bombardier Canadair DSD

CSA Engineering, Inc.

Engineering Software Research & Development Inc.

FAA Technical Center

Fatigue Concepts

Fatigue Technology Inc.

Jentek Sensors Inc.

Karta Technology Inc.

Knowledge Systems Inc.

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Martec Limited

Measurement Systems Inc.

Metal Improvement Co., Inc.

Moog Inc.

MTS Systems Corp.

National Technical Systems

Naval Air Systems Command

Nondestructive Testing Information Analysis Center (NTIAC)

Oak Ridge National Laboratory

RADA Electronic Industries Ltd.

R/D Tech

Southwest Research Institute

Systems & Electronics, Inc.

USAF/Boeing/Digital Wave Corp.

INTRODUCTION

This report contains the proceedings of the 1997 USAF Structural Integrity Program Conference held at the Hyatt Regency San Antonio in San Antonio, Texas, from 2-4 December 1997. The Conference, which was sponsored by the Aeronautical Systems Center's Engineering Directorate, the Air Force Research Laboratory's Materials and Manufacturing Directorate and Air Vehicles Directorate, was hosted and co-sponsored by the San Antonio Air Logistics Center Aircraft Directorate, Aircraft Structural Integrity Branch. This conference, as in previous years, was held to permit experts in the field of structural integrity to communicate with each other and to exchange views on how to improve the structural integrity of military weapon systems and commercial aircraft. Sessions were primarily focused on life enhancement, bonded composite repairs, NDE/I, fatigue and crack growth, analytical methods, dynamics and force management. This year, as in previous years, our friends from outside the U.S. borders provided the audience with outstanding presentations on activities within their countries. It is anticipated that this conference will continue to include their contributions in future years. This year 17 countries with 83 foreign attendees were represented in the audience.

The sponsors are indebted to their hosts for their support of the conference. The sponsors are also indebted to the speakers for their contributions. In particular, thanks are due to the three luncheon speakers for their informative presentations, Mr. C. Tiffany on The National Research Council Report on Aging of US Air Force Aircraft; Captain J. Leclerc on A Risk Based Approach to the Management of Critical Areas on the IFOSTP FT-55 Full-Scale Fatigue Test Article; and Mr. J. Morgan on the B-1B Horizontal Stabilizer Substructure Failures.

As usual, much of the success of the Conference is due to the efforts of Jill Jennewine and her staff, including Esther Burnett and Kandi Granato, from Universal Technology Corporation. Their cooperation is greatly appreciated.

JOHN W. LINCOLN	GARY K. WAGGONER	JIM RUDD	JIMMY TURNER
ASC/ENF	AFRL/MLS	AFRL/VAB	SA-ALC/LADD

SESSION I

OVERVIEWS

Chairman - *J. Rudd*
Air Force Research Laboratory

Economic Implications of DADTA for Military Aircraft Operators of Non-USAF Fleets

by

Kurt Schrader, Group Leader

Hal Burnside, Director

William Sparks, Consultant

Joe Cardinal, Group Leader

Aerospace & Reliability Engineering Department

Southwest Research Institute

Presentation Topics

- Overview of the Elements of a DADTA Program
- Observations on DADTA for foreign military operators of USAF aircraft
- Illustrations of Risk and Cost Issues Associated with a DADTA
- Observations and Conclusions

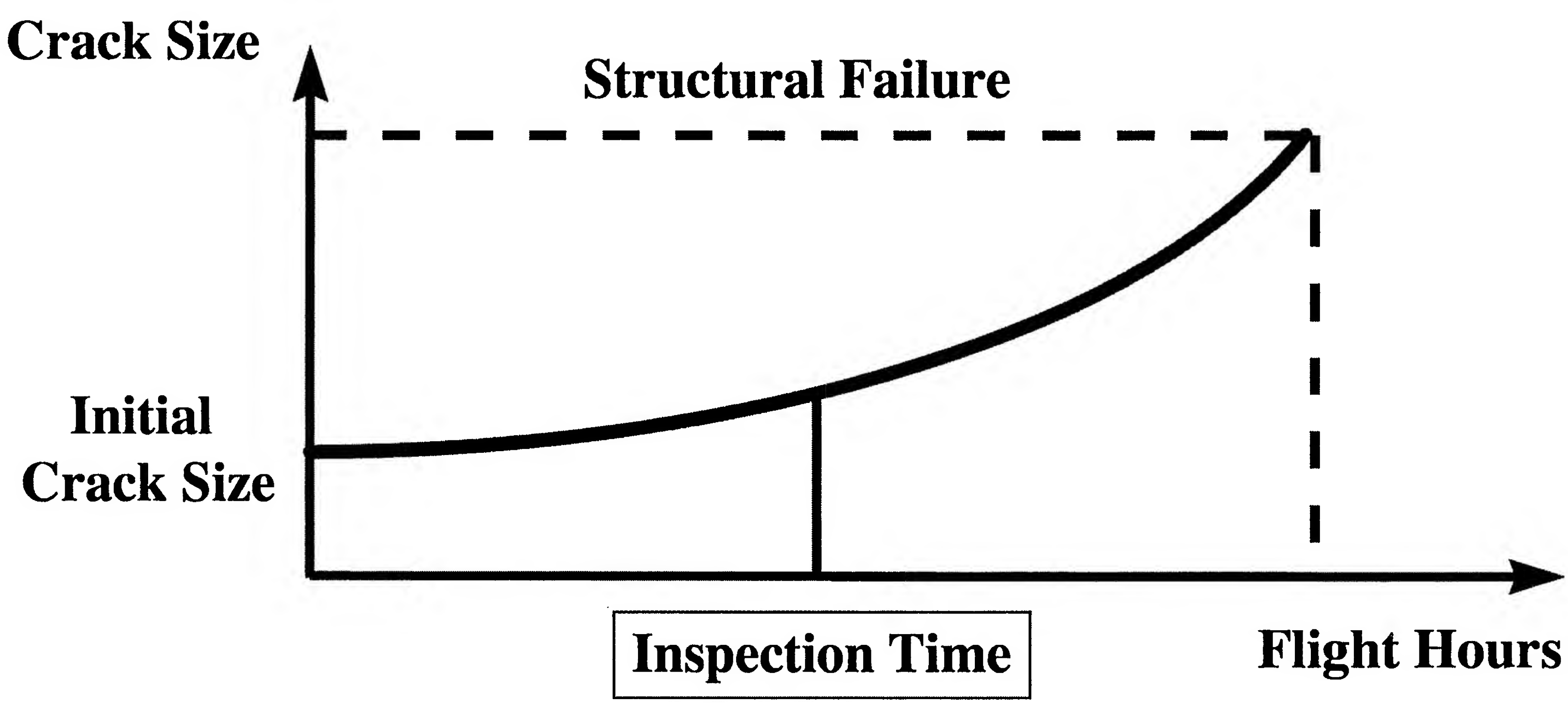
Overview

Elements of a DADTA Program

- Individual country usage assessment
- Stress spectra development for each fatigue critical location (FCL)
- Laboratory material and coupon spectra testing
- Fracture mechanics analysis of FCLs
- Determination of initial and recurring inspection times
- Implementation of inspection requirements

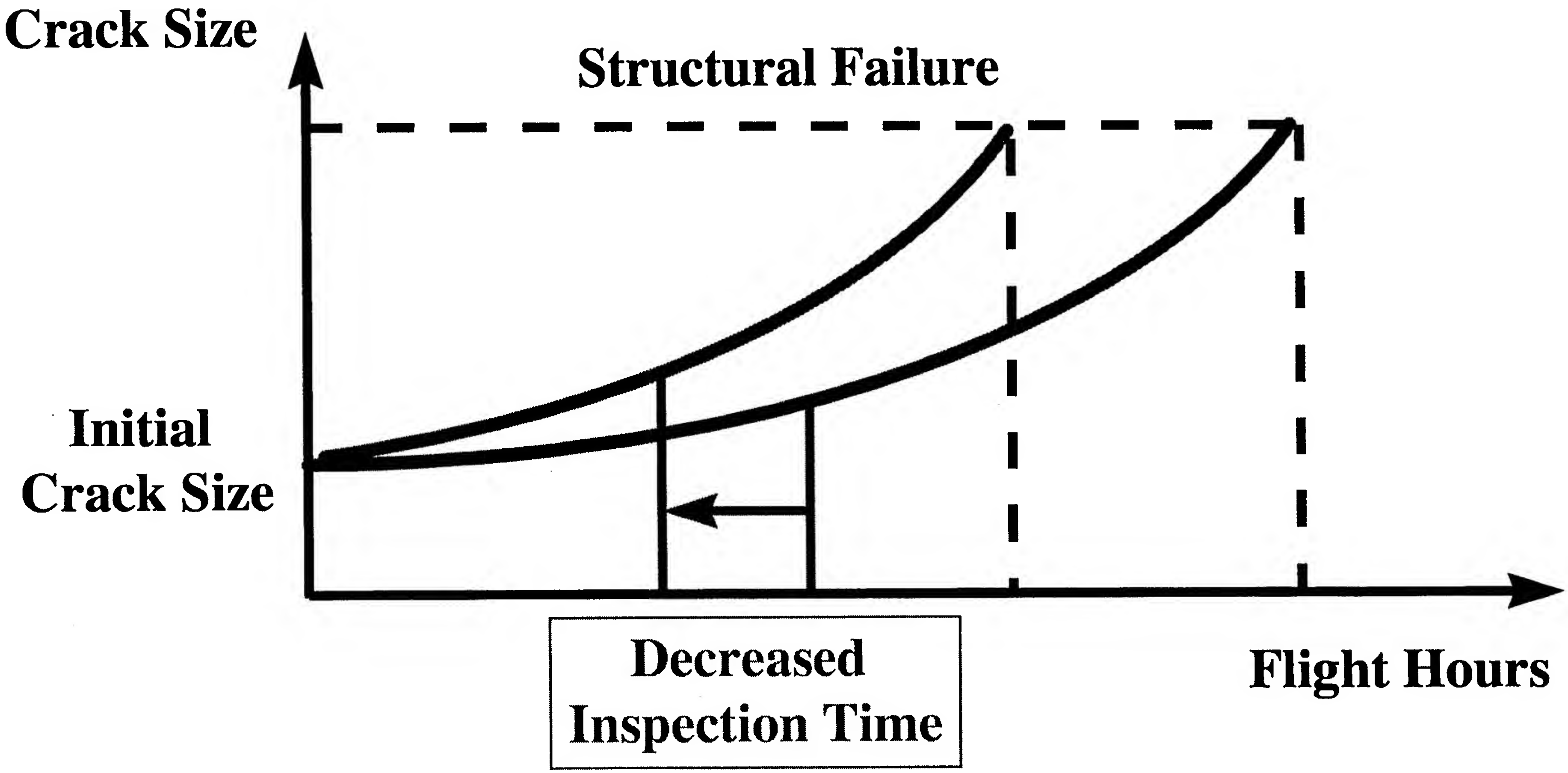
DADTA Overview

USAF Requires That an Inspection Be Made at 1/2 the Time Required for a Crack to Reach Critical Size



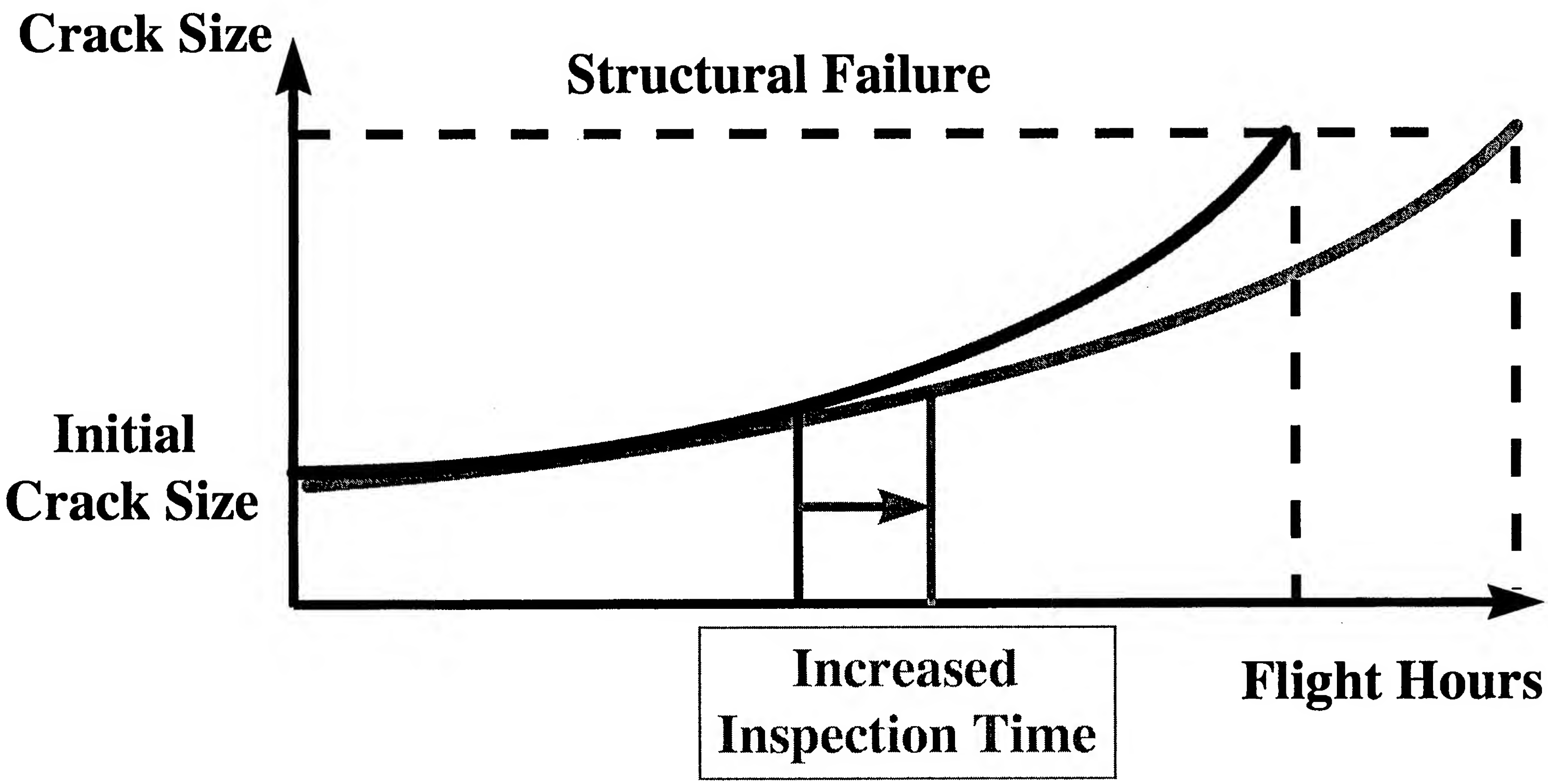
DADTA Overview

More Severe Usage Decreases Inspection Times



DADTA Overview

Less Severe Usage Increases Inspection Times



DADTA Observations

FMS Operators of USAF Aircraft

- FMS users generally follow USAF lead on structural inspections
- Recent examples where USAF fleet was not representative of FMS fleet
  - USAF F-5 Drawdown -- FMS users no longer have a common fleet leader
  - USAF T-37 Structural Life Extension -- FMS users no longer have common configuration with USAF

DADTA Observations

FMS Operators of USAF Aircraft (continued)

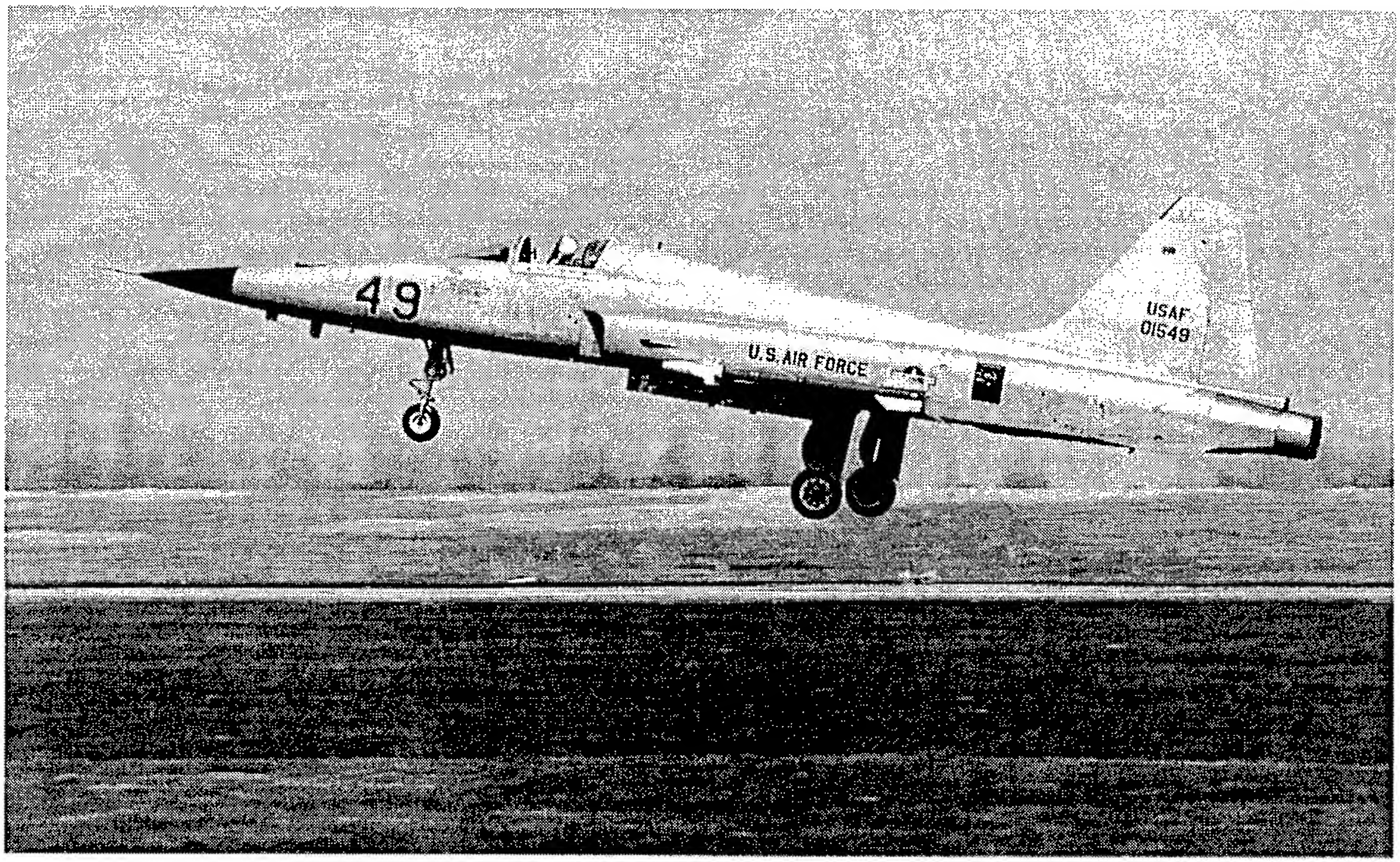
- Lack of a specific DADTA is often not perceived as a safety of flight risk
  - usage differences
  - structural variations
- Initial costs of a DADTA is often considered a "luxury" relative to other operational costs, especially for operators of small fleets

An Example

Quantify the Risk and Costs Associated with a DADTA

- Results of recently completed DADTAs for three F-5E FMS users
- Results of original DADTA for USAF F-5E Security Assistance Program (SAP) usage
- USAF *PROF* computer program

F-5E

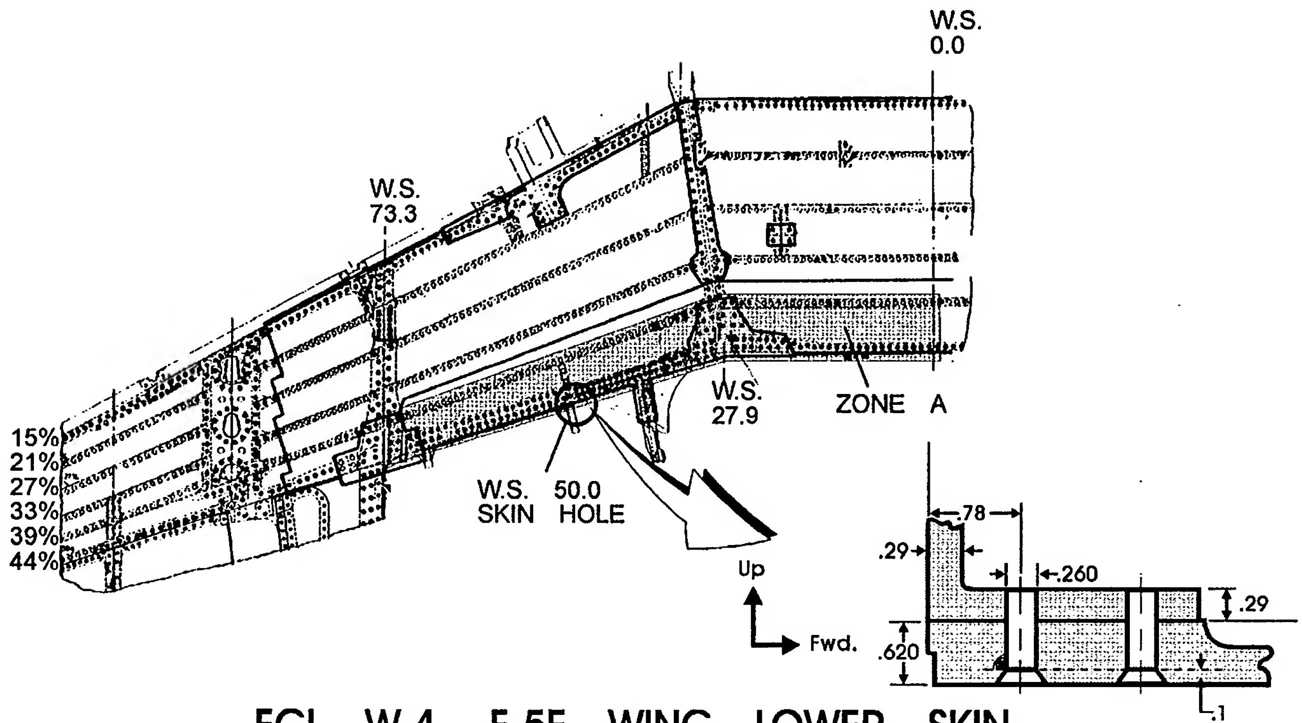


An Example

Quantify the Risk and Costs Associated with a DADTA (continued)

- Focus will be on a single location on the wing
  - FCL W-4, lower skin fastener hole at Wing Station 50 and the 44% spar
    - coldworked
    - countersunk
- W-4 sets inspection interval for approximately 400 fastener holes on 39% and 44% spars

FCL W-4, F-5E Wing Lower Skin



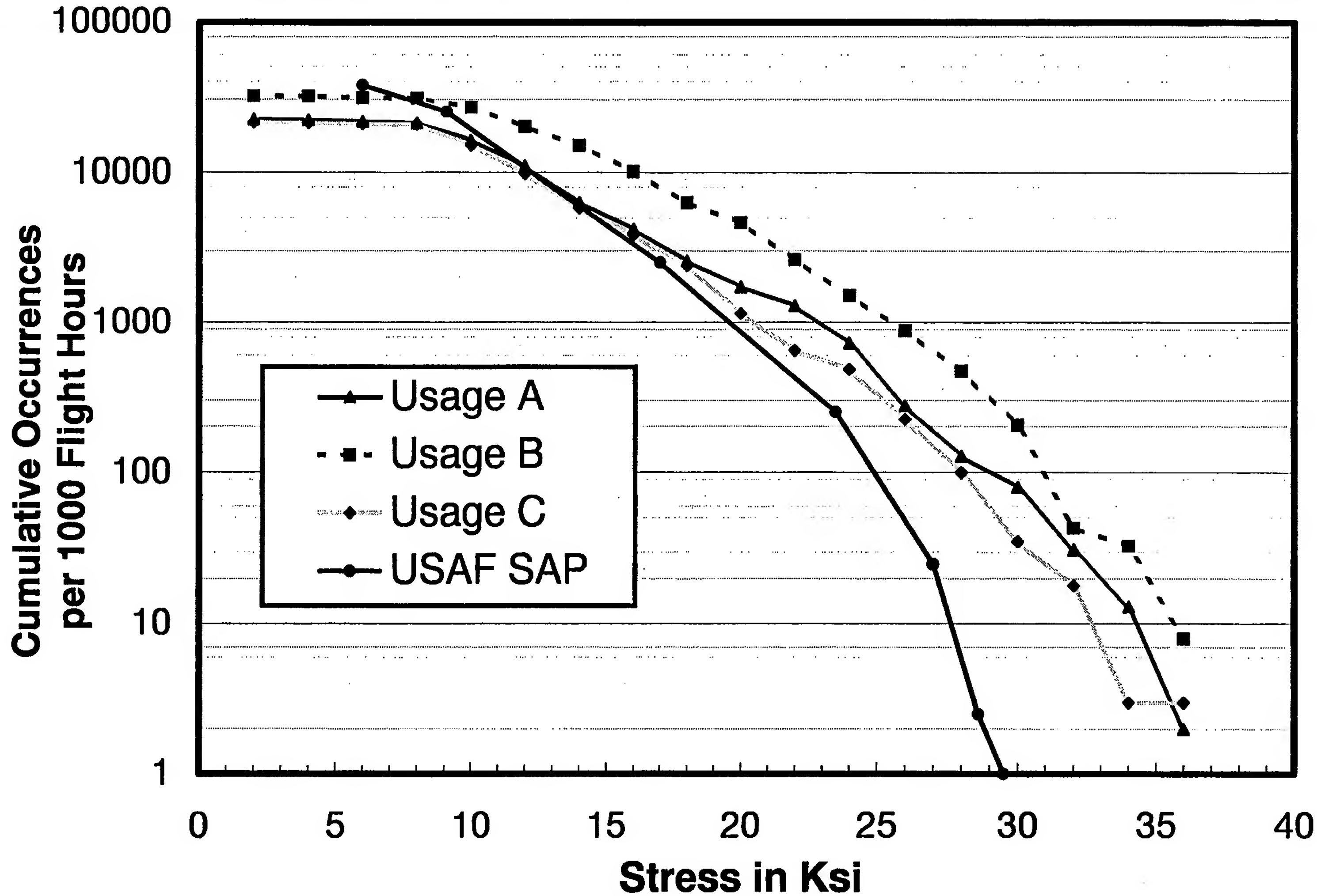
FCL W-4, F-5E WING LOWER SKIN

DADTA Overview

DADTA Summary Information

- Stress Spectra at FCL W-4
- Crack Growth Retardation
- Crack Growth Curves at FCL W-4
- Inspection Times

Stress Spectra for Wing FCL W-4



DADTA for Military Aircraft Operators of Non-USAF Fleets

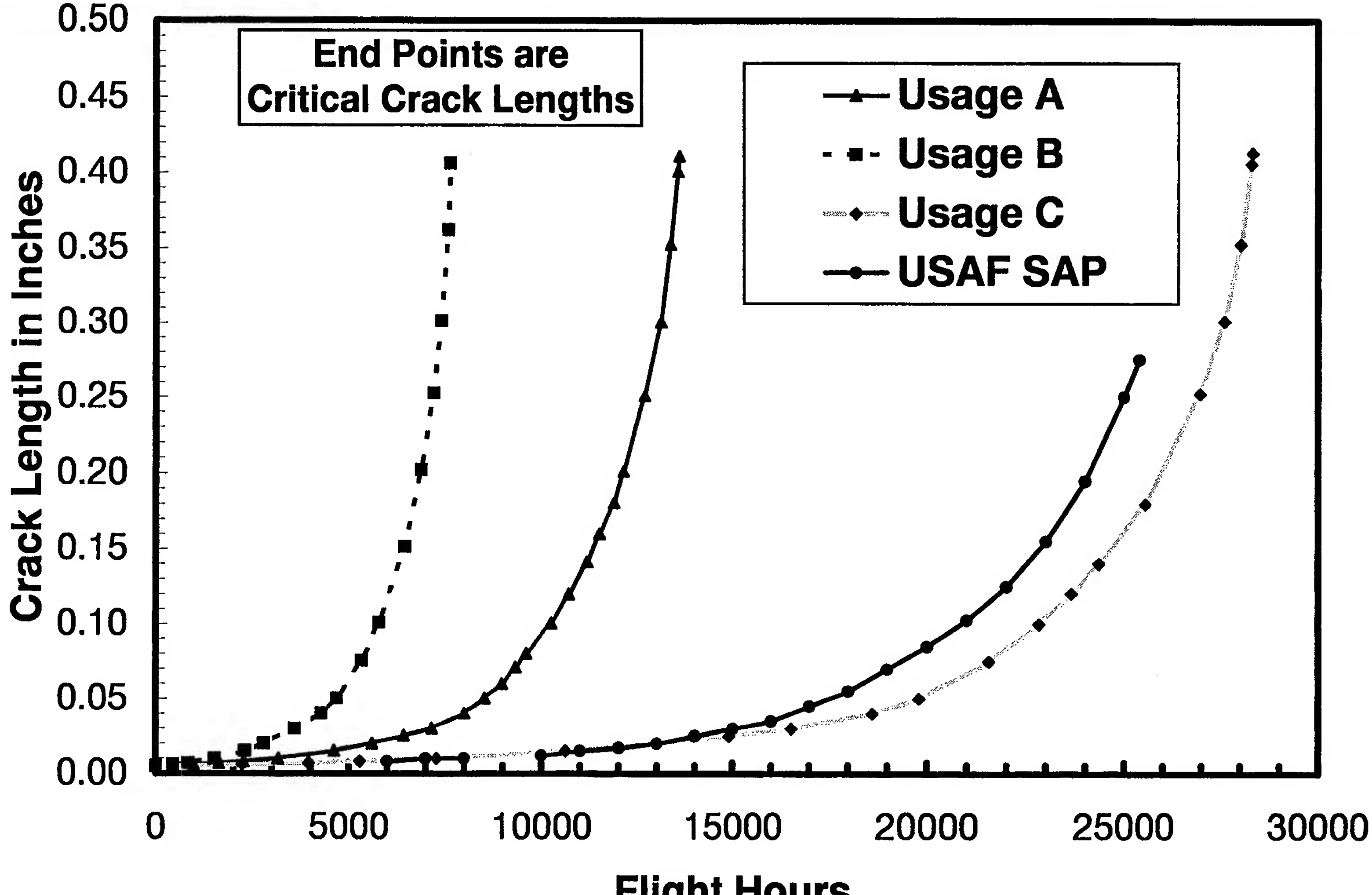
1997 USAF ASIP Conference

Crack Growth Retardation

Obtained from Coupon Spectrum Tests

Usage	Modified Willenborg Shut-Off Ratio
A	2.1
B	1.9
C	1.9
USAF SAP	1.9

Crack Growth Curves



Initial and Recurring Inspection Times

Aircraft Usage	Initial Inspection (Flight Hours)	Recurring Interval (Flight Hours)
A	6500	1400
B	3600	800
C	13500	2300
USAF SAP	12700	2300

Probability of Fracture Calculations

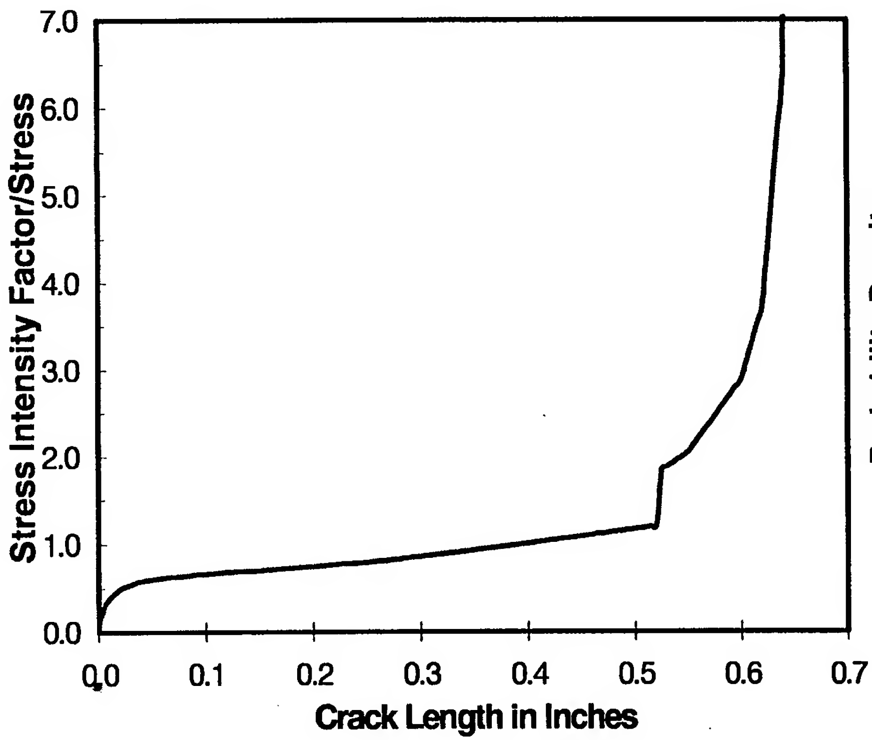
Probability Calculations Using ***PROF***

- **Pr**obability **O**f **F**racture
- Written by the University of Dayton Research Institute for Wright Labs in 1991
- Input Data
  - Materials and Geometry
  - Aircraft and Usage
  - Inspection and Repair

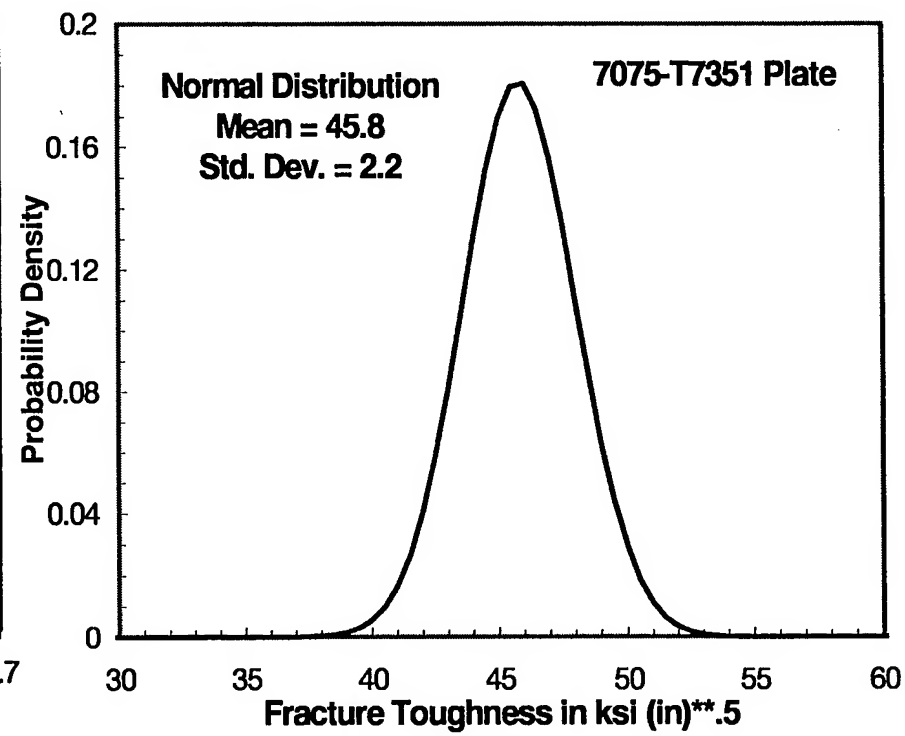
Inputs to PROF

Materials and Geometry

Crack Stress Intensity Factor



Distribution of Fracture Toughness

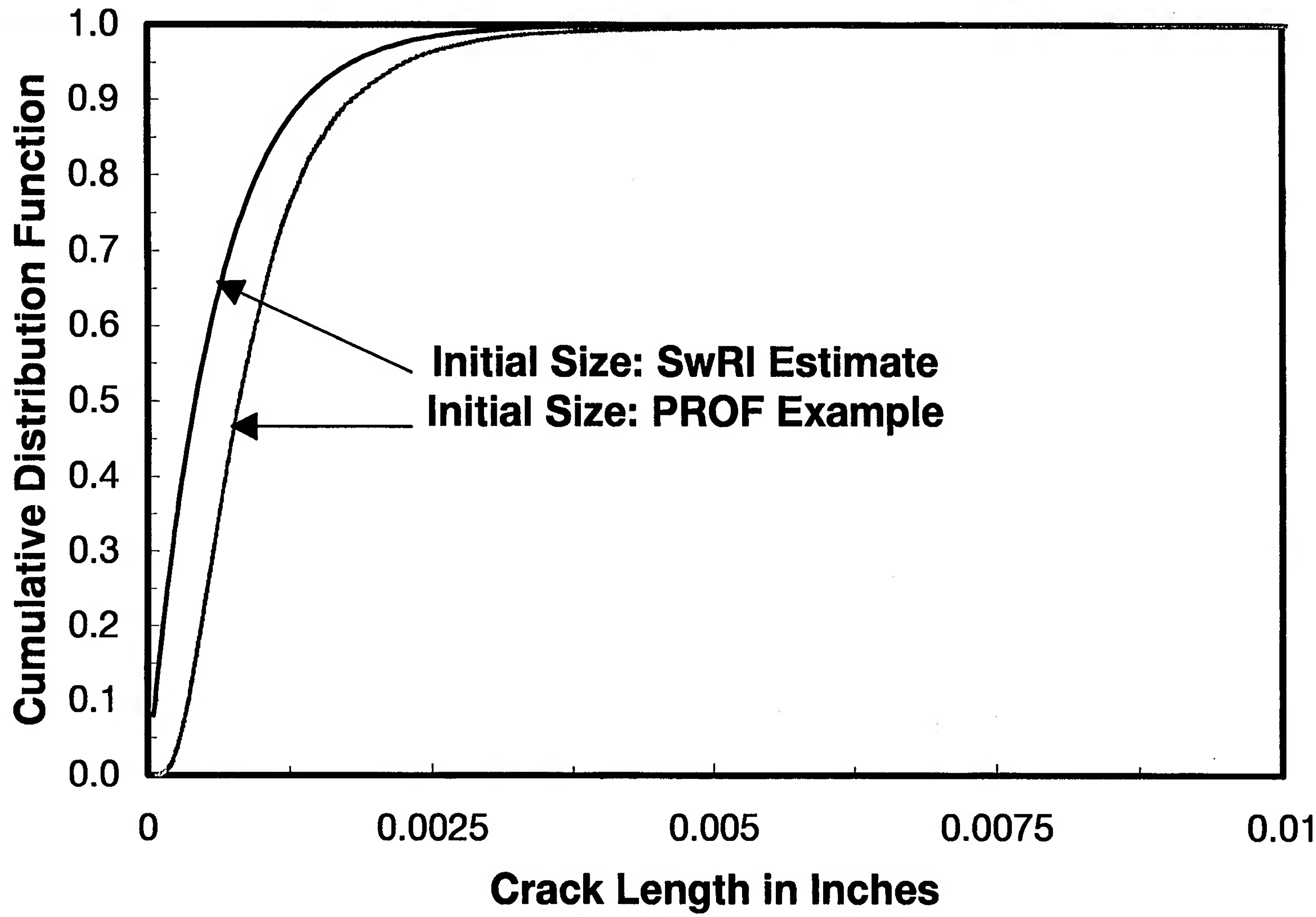


Inputs to PROF

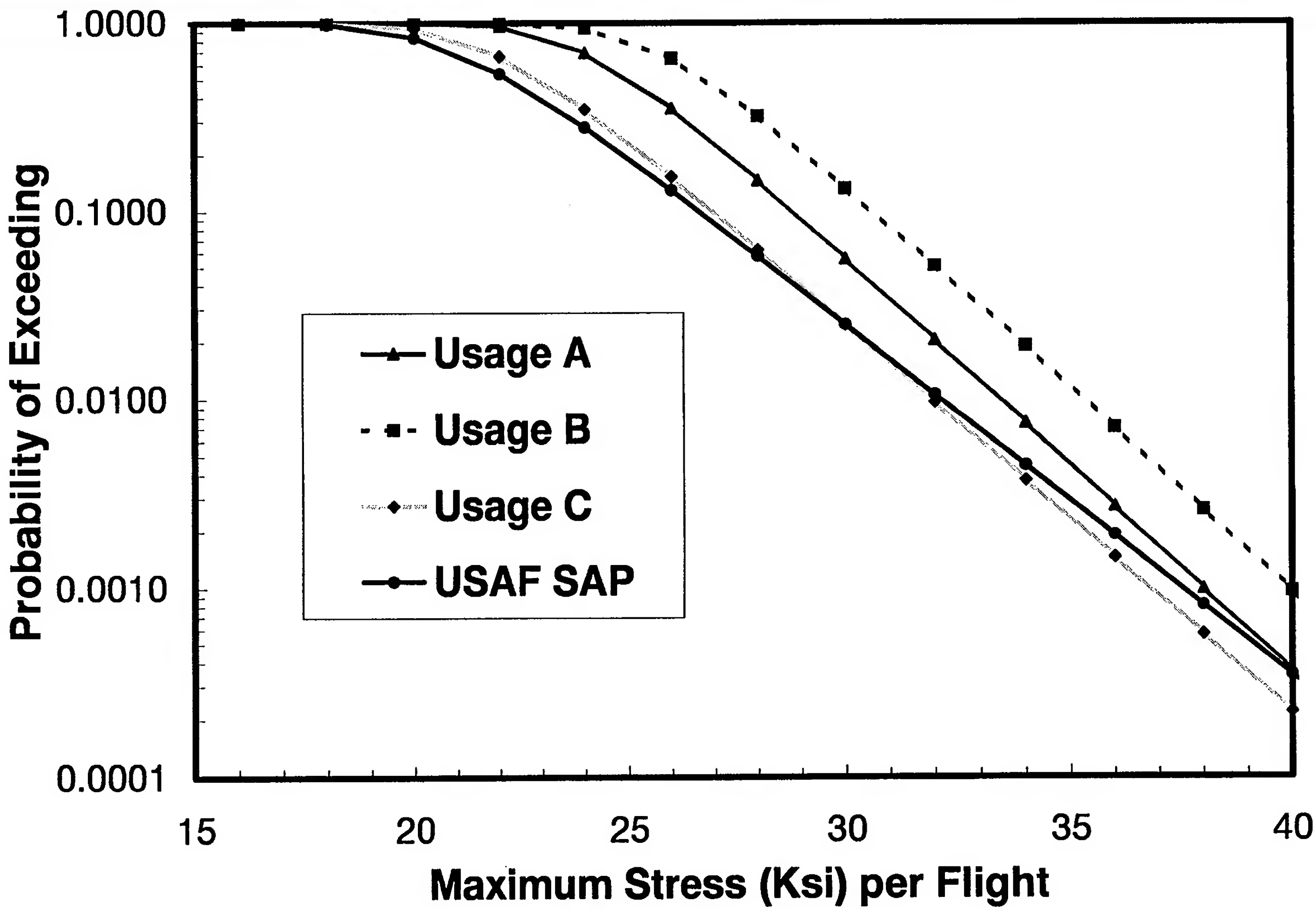
Aircraft and Usage

- Number of analysis locations per airframe and size of fleet
- Crack size distribution at start of analysis
- Damage tolerance analysis (DTA) crack growth curve
- Extreme value distribution of maximum stress per flight

Initial Crack Size Distribution



Extreme Value Distribution of Stress Spectra



DADTA for Military Aircraft Operators of Non-USAF Fleets

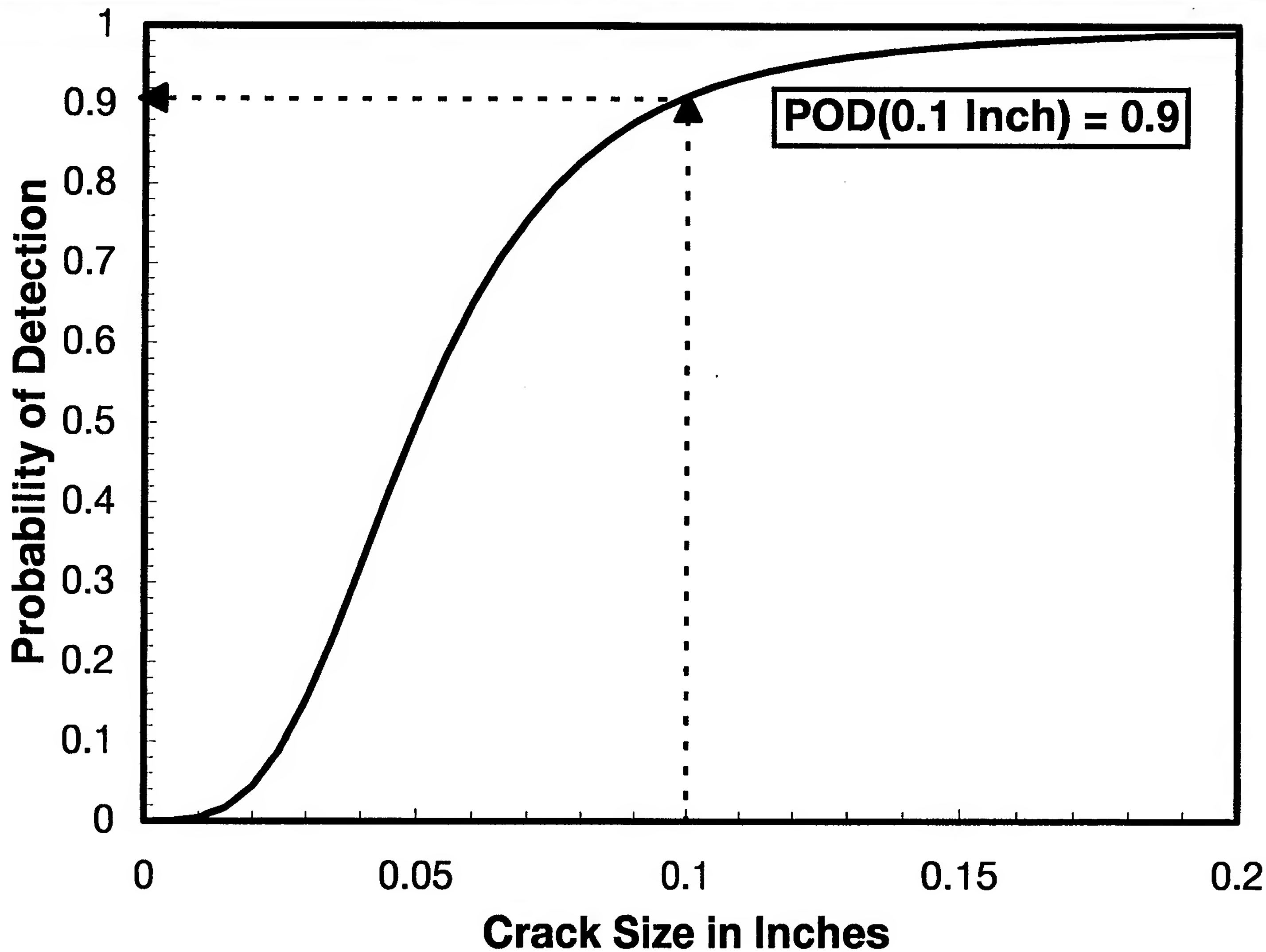
1997 USAF ASIP Conference

Inputs to PROF

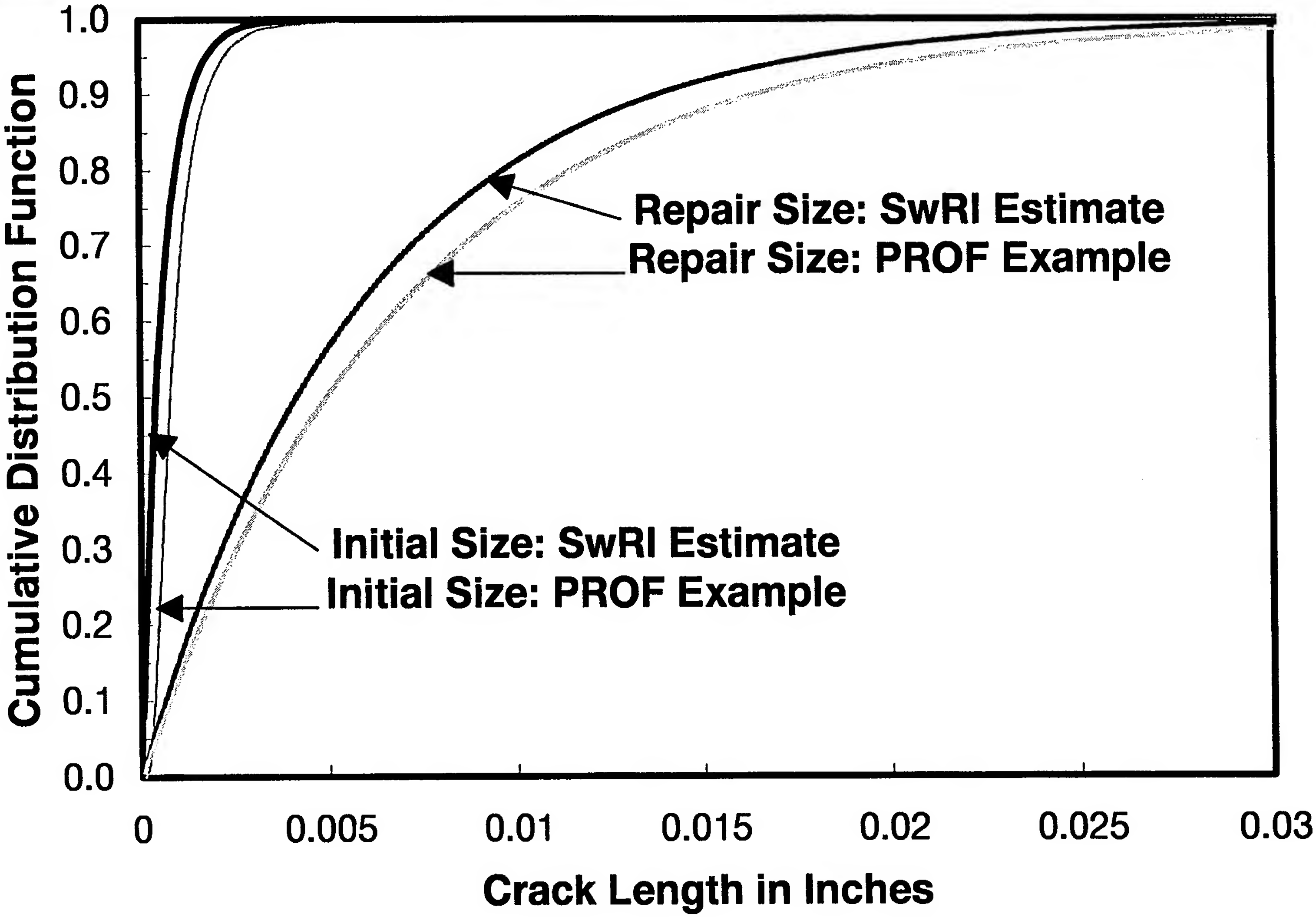
Inspection and Repair

- Initial and recurring inspection times
- Cumulative lognormal probability of detection (POD) for NDI
- Crack size distribution of repaired crack sites

Probability of Crack Detection



Initial and Repair Crack Size Distribution



Probability of Fracture (POF) Calculations

$POF_E(t)$ = Probability of fracture for an element (FCL)

K = Number of equivalent FCLs in the aircraft

N = Number of aircraft in fleet

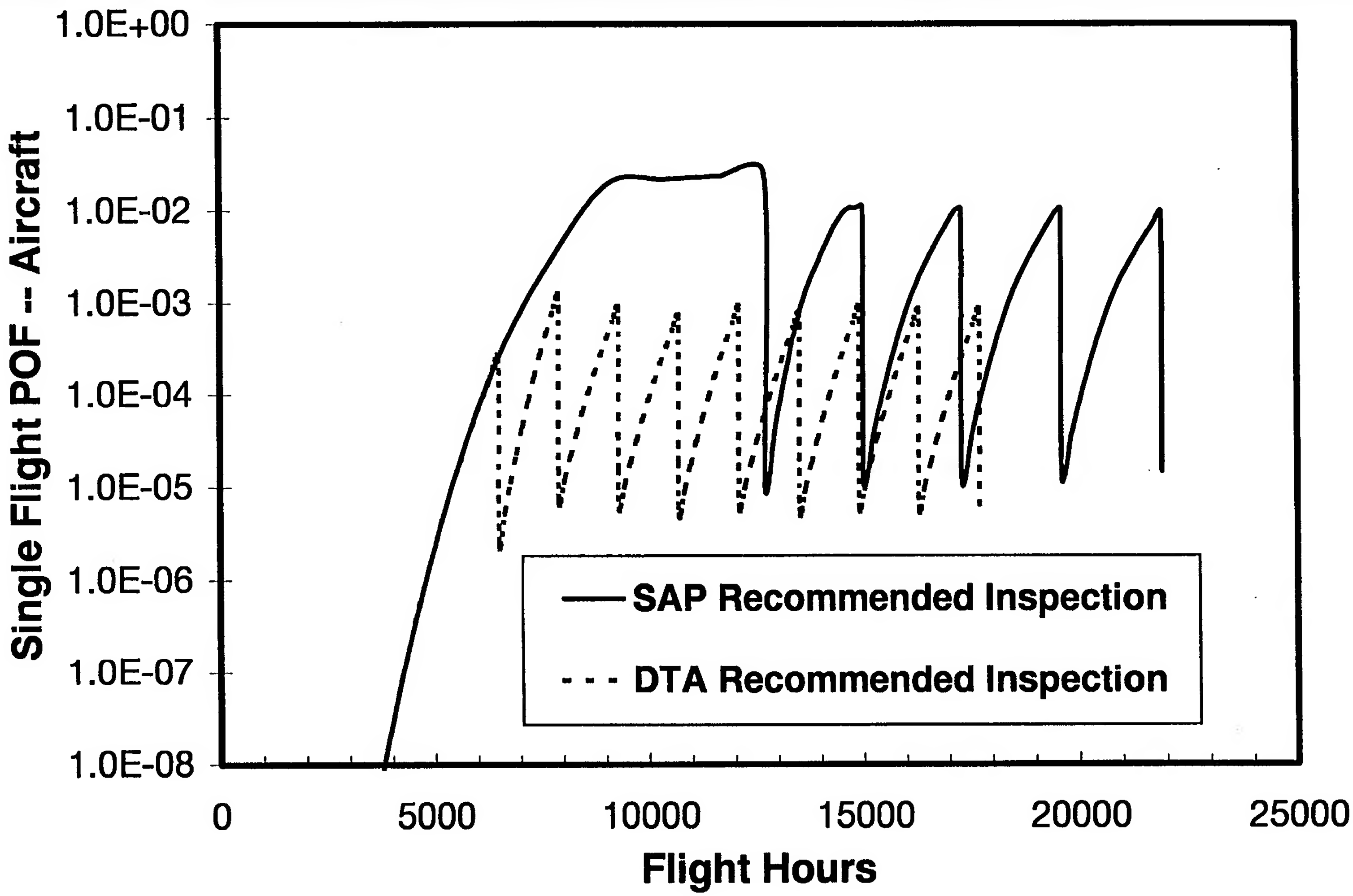
Single Aircraft Probability of Fracture

$$POF_A(t) = K \times POF_E(t)$$

Fleet Probability of Fracture

$$POF_F(t) = N \times POF_A(t)$$

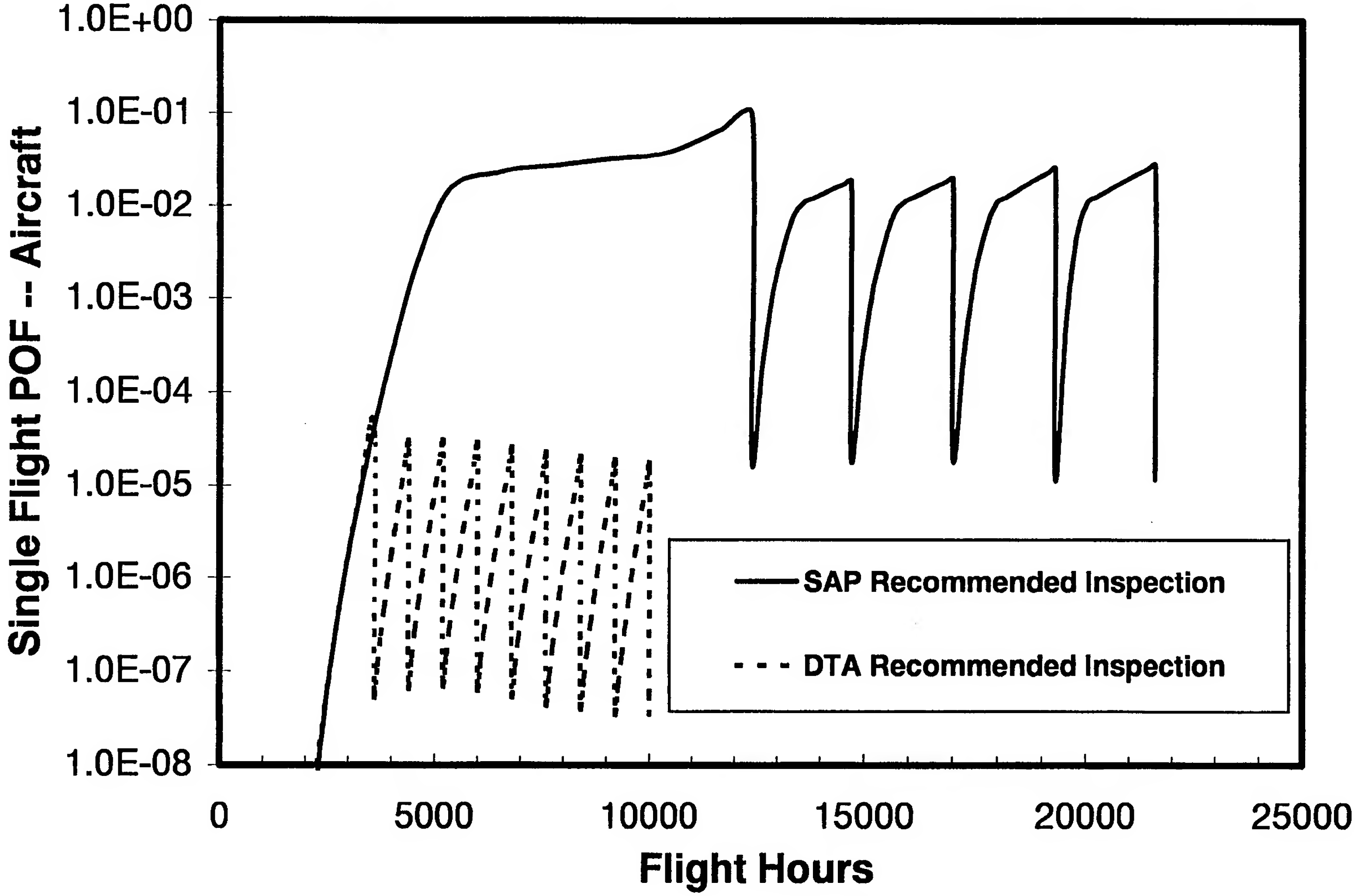
Usage A Probability of Fracture



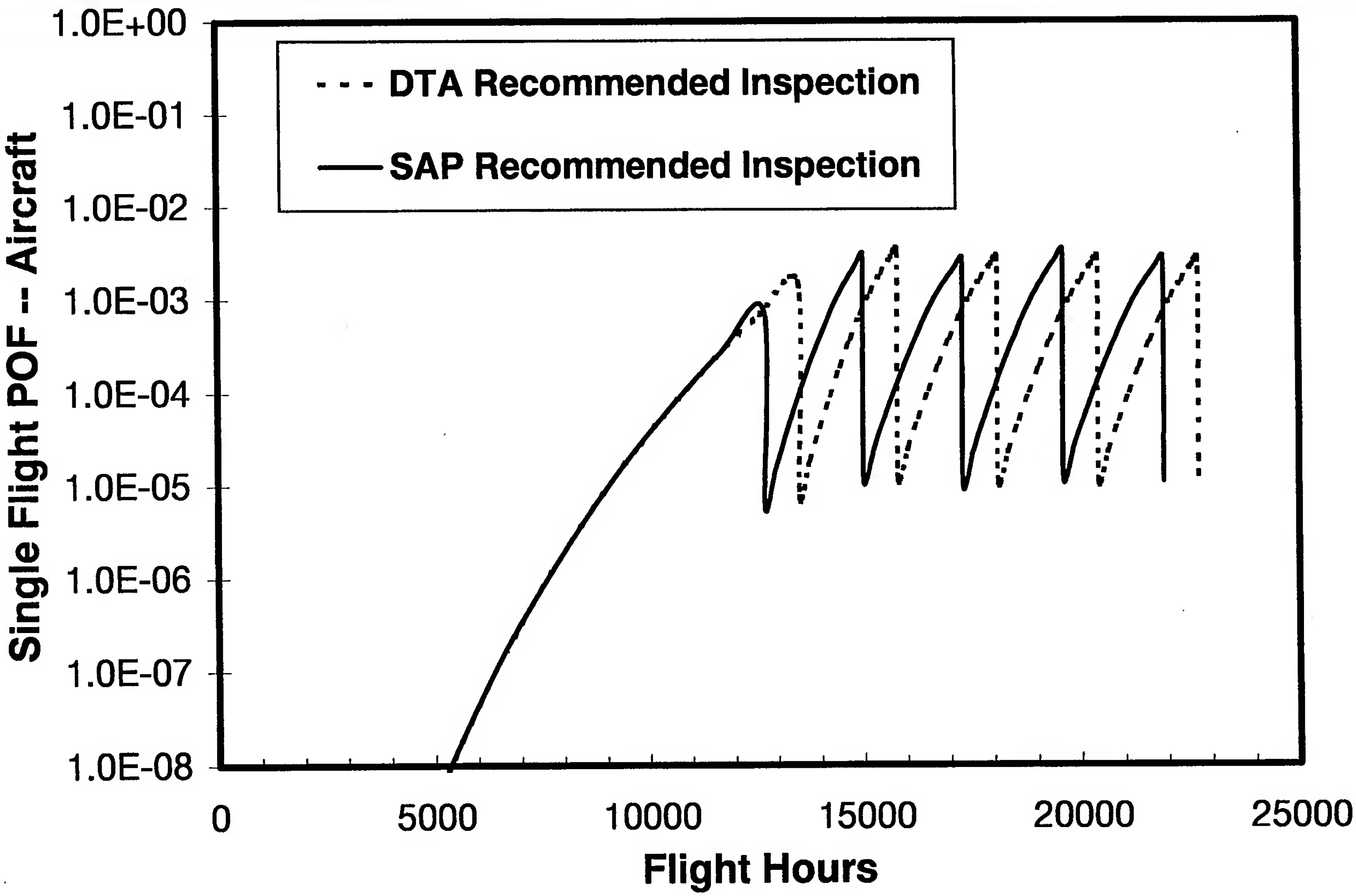
DADTA for Military Aircraft Operators of Non-USAF Fleets

1997 USAF ASIP Conference

Usage B Probability of Fracture



Usage C Probability of Fracture



Risk Reduction by Performing DADTA

Aircraft Usage	Initial Inspection (Flight Hours)	Recurring Interval (Flight Hours)	Risk Reduction (Orders of Magnitude)
A	6500	1400	1 - 2
B	3600	800	2 - 3
C	13500	2300	None
USAF SAP	12700	2300	—

Expected Maintenance Costs

Expected Cost of Fracture, Inspection and Repair in Each Usage Interval

$$
\begin{aligned}
E(c) &= K \times N \times C_I && \text{(Inspection)} \\
&+ K \times N \times P(\Delta a_1) \times C_{R1} && \text{(Repair)} \\
&+ K \times N \times P(\Delta a_2) \times C_{R2} && \text{(Replacement)} \\
&+ POF_A(t) \times N \times C_F && \text{(Fracture/Aircraft Loss)}
\end{aligned}
$$

Expected Maintenance Costs

Terminology

$E(c)$ = Expected cost in each usage interval

$POF_A(t)$ = POF for single aircraft

$P(\Delta a_i)$ = Proportion of detected cracks in interval Δa_i

K = Number of repair details

N = Number of aircraft in fleet

C_I = Cost of Inspection

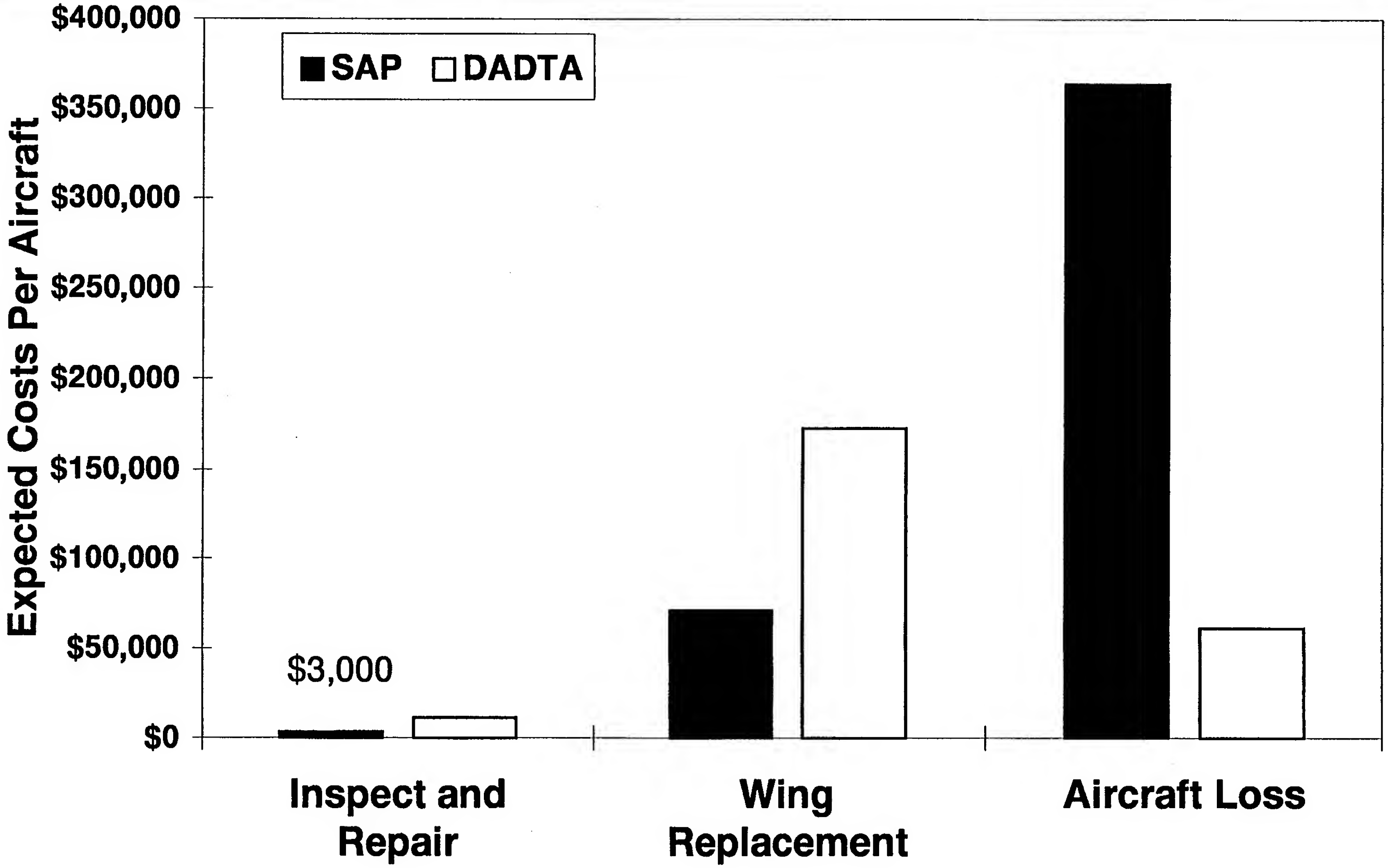
C_{Ri} = Cost of Repair/Replacement in interval Δa_i

C_F = Cost of Fracture

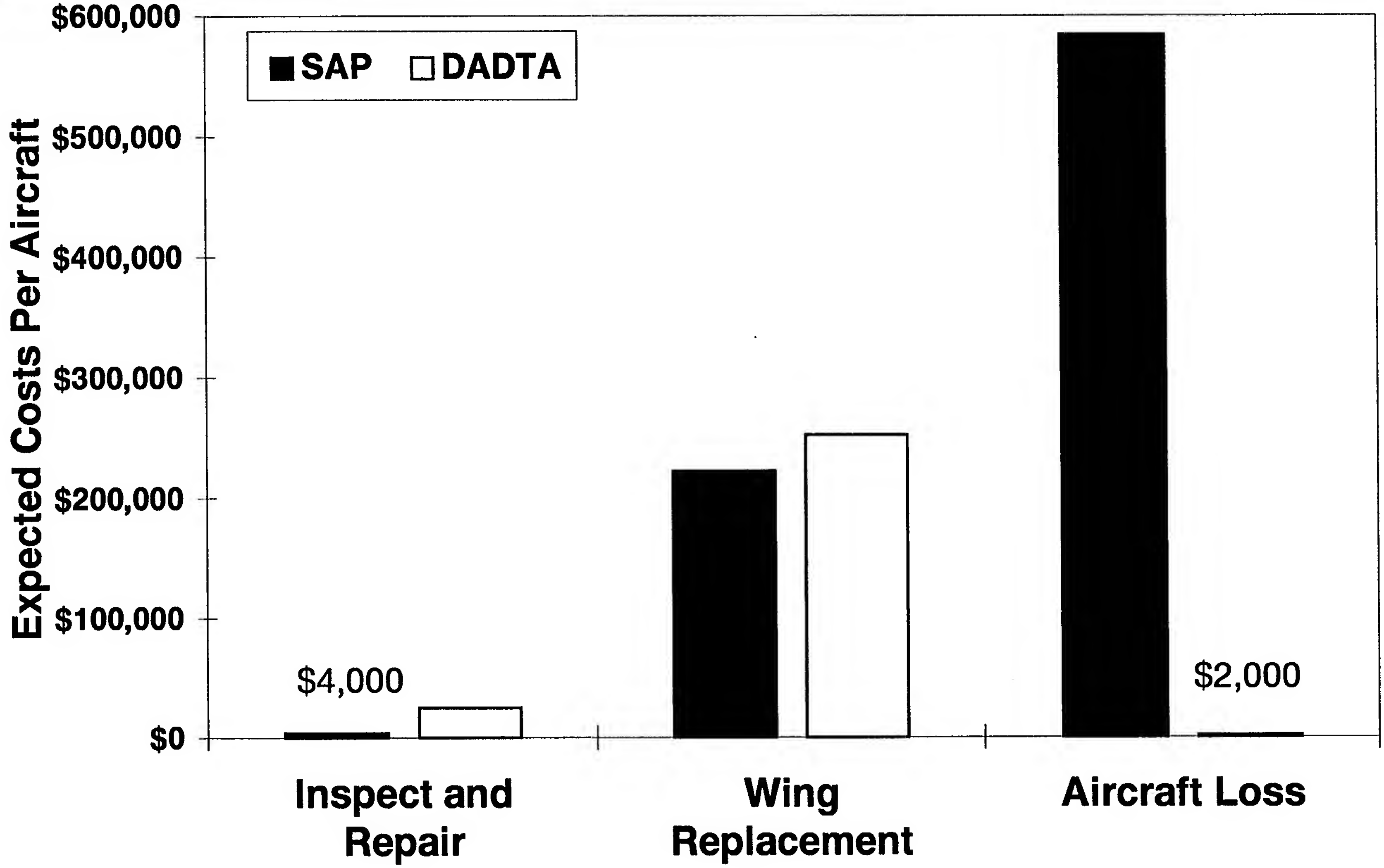
Cost Assumptions

Labor rate per hour:	\$100
NDI Inspection per wing: 16 hours	\$1,600
Repair (0.03" radial crack): 5 hours	\$500
Engineering/repair (0.03" < c < 0.1" crack): 50 hours	\$5,000
Wing replacement (c > 0.1" crack):	\$1,000,000
Aircraft loss - Production Configuration:	\$5,000,000
Aircraft loss - Post Avionics Upgrade:	\$10,000,000
Cost of Durability and Damage Tolerance Analysis:	\$1,000,000

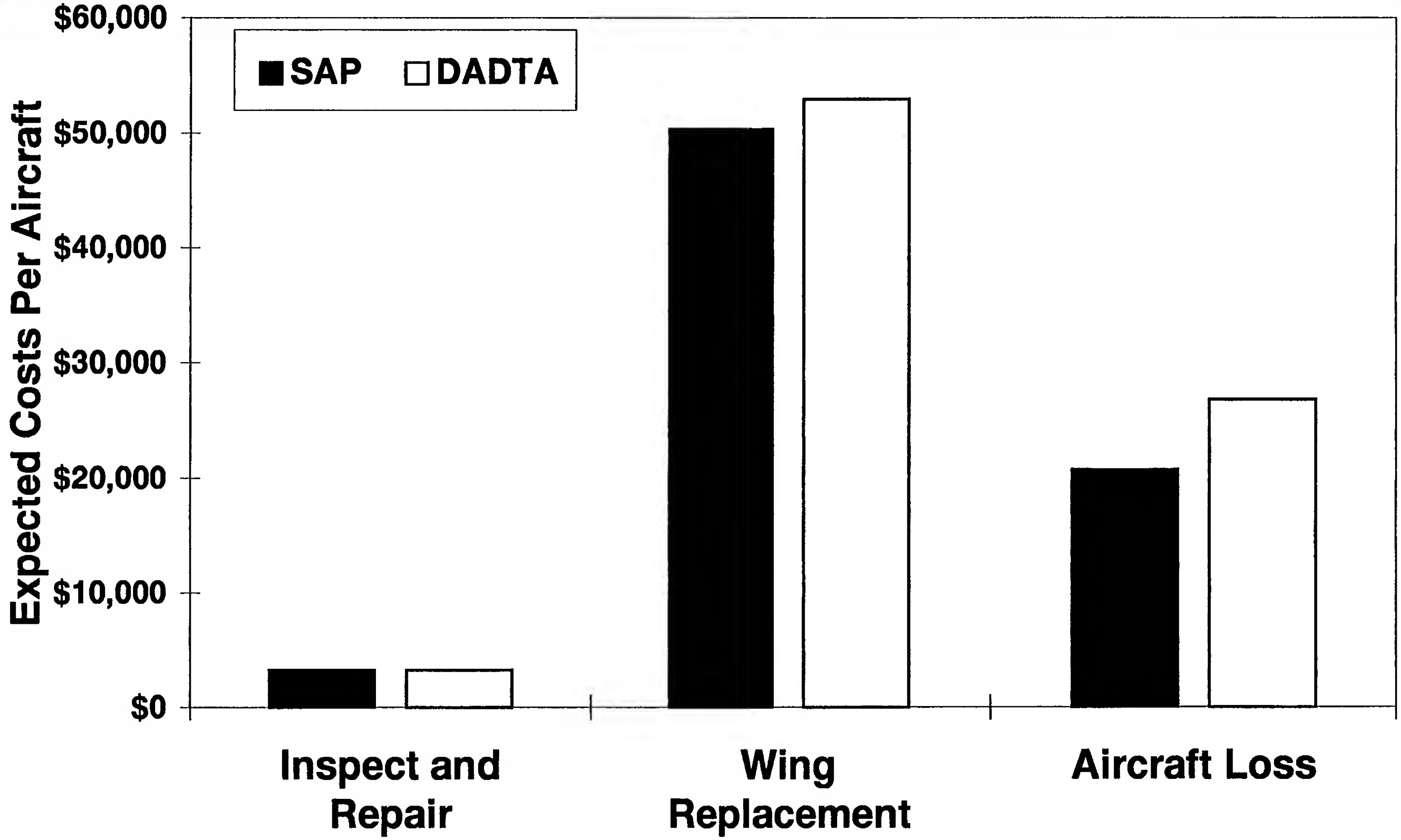
Costs at 15,000 Flight Hours - Usage A



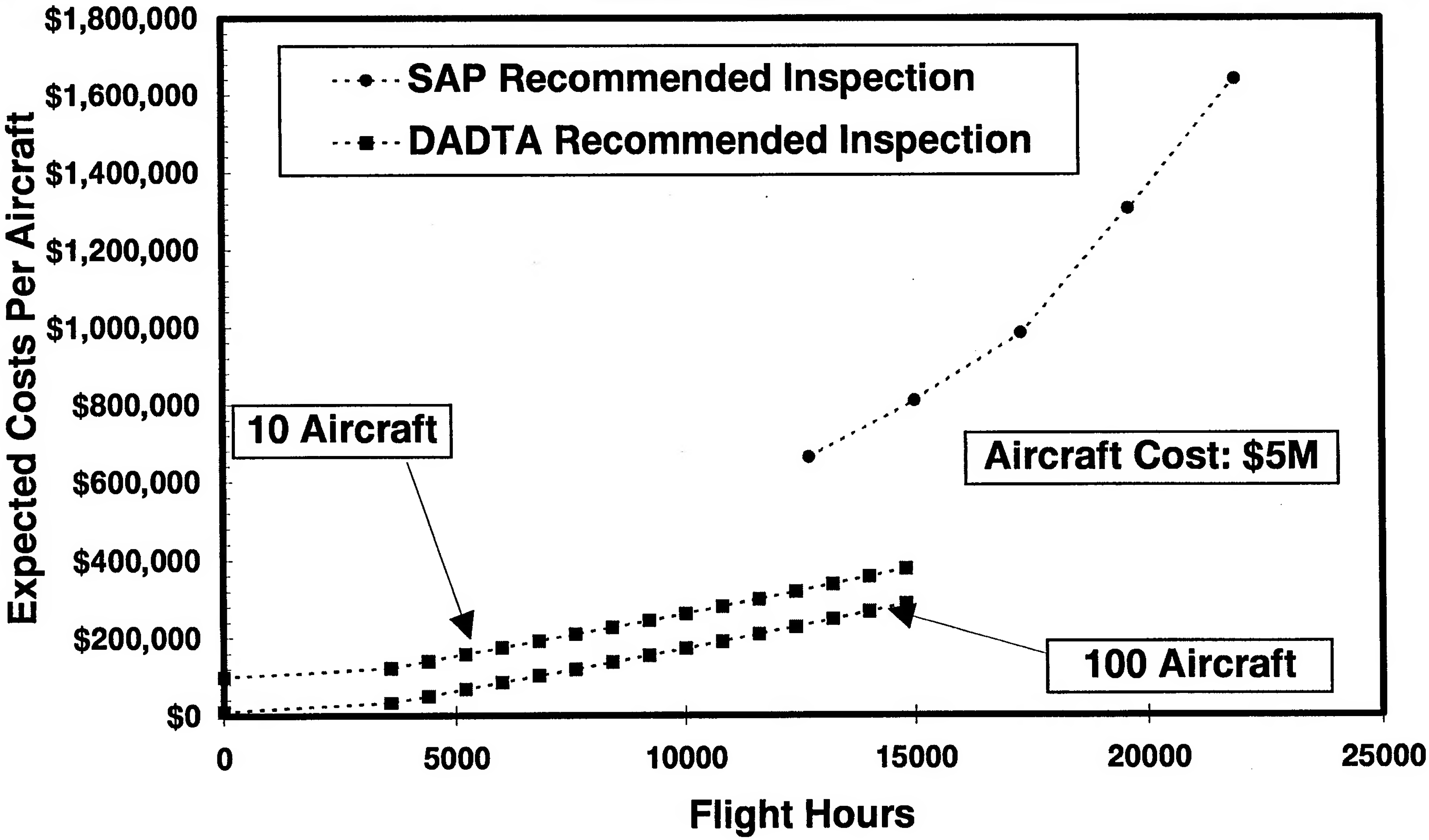
Costs at 15,000 Flight Hours - Usage B



Costs at 15,000 Flight Hours - Usage C



Total Costs (Including DADTA) - Usage B



DADTA for Military Aircraft Operators of Non-USAF Fleets

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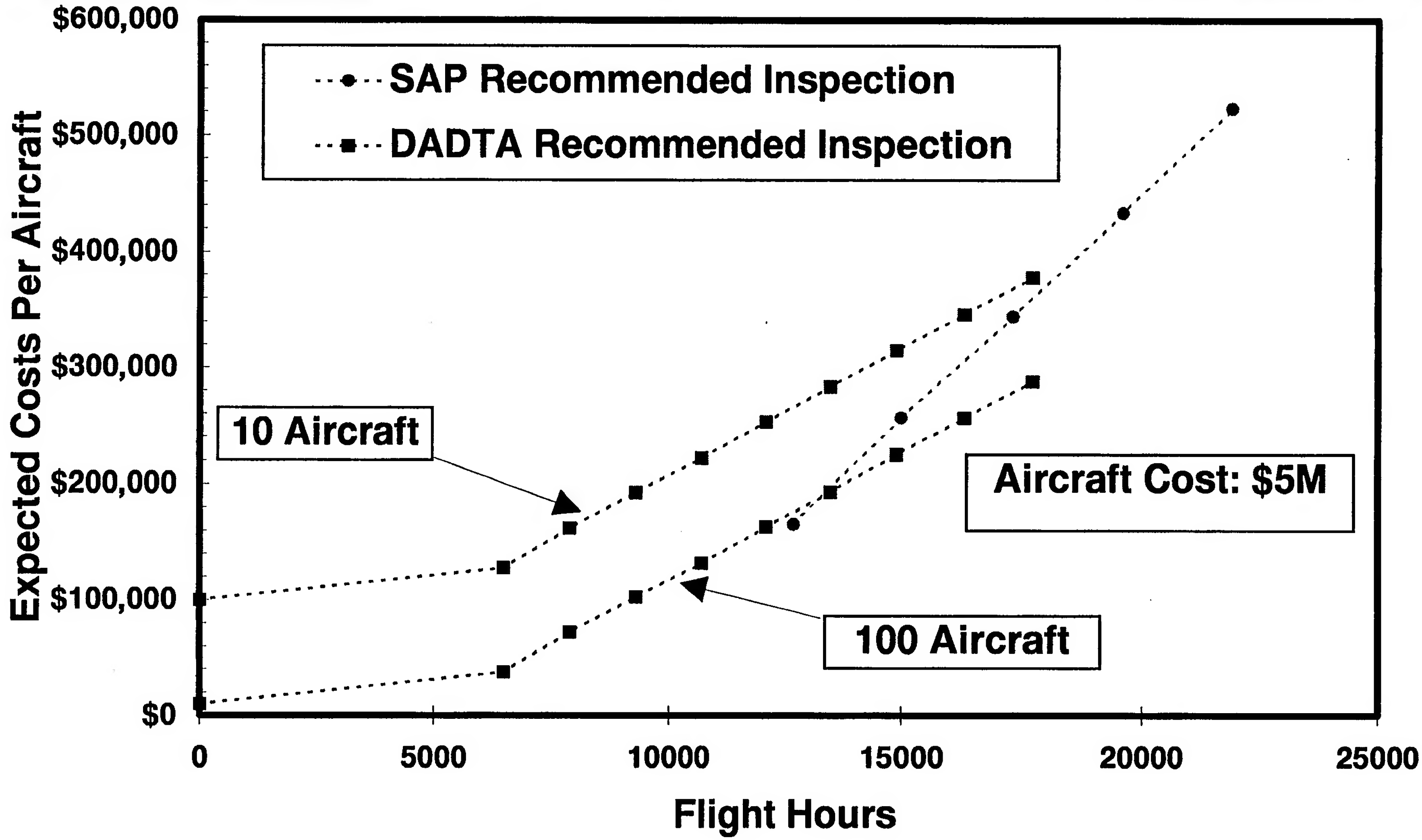
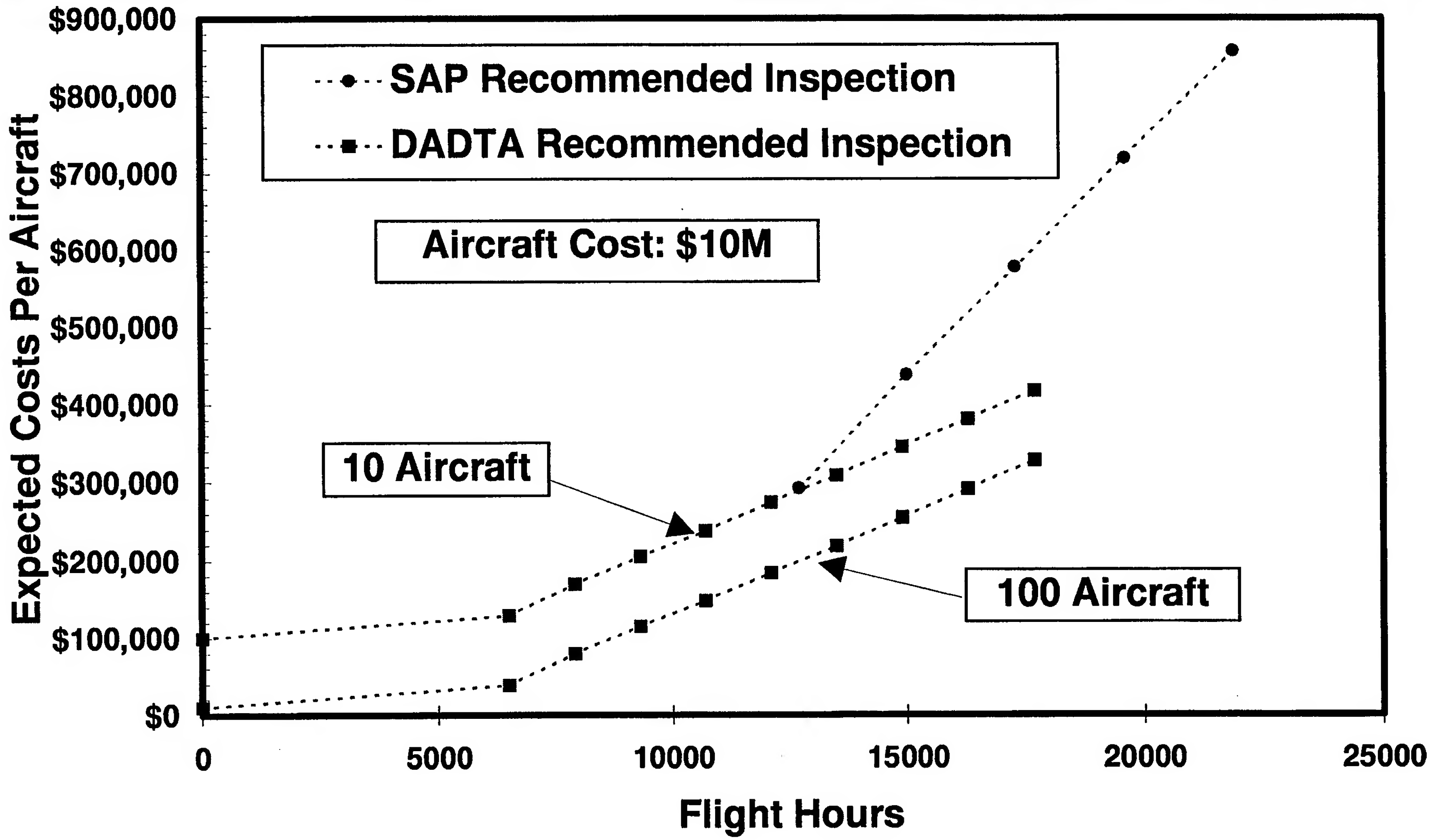
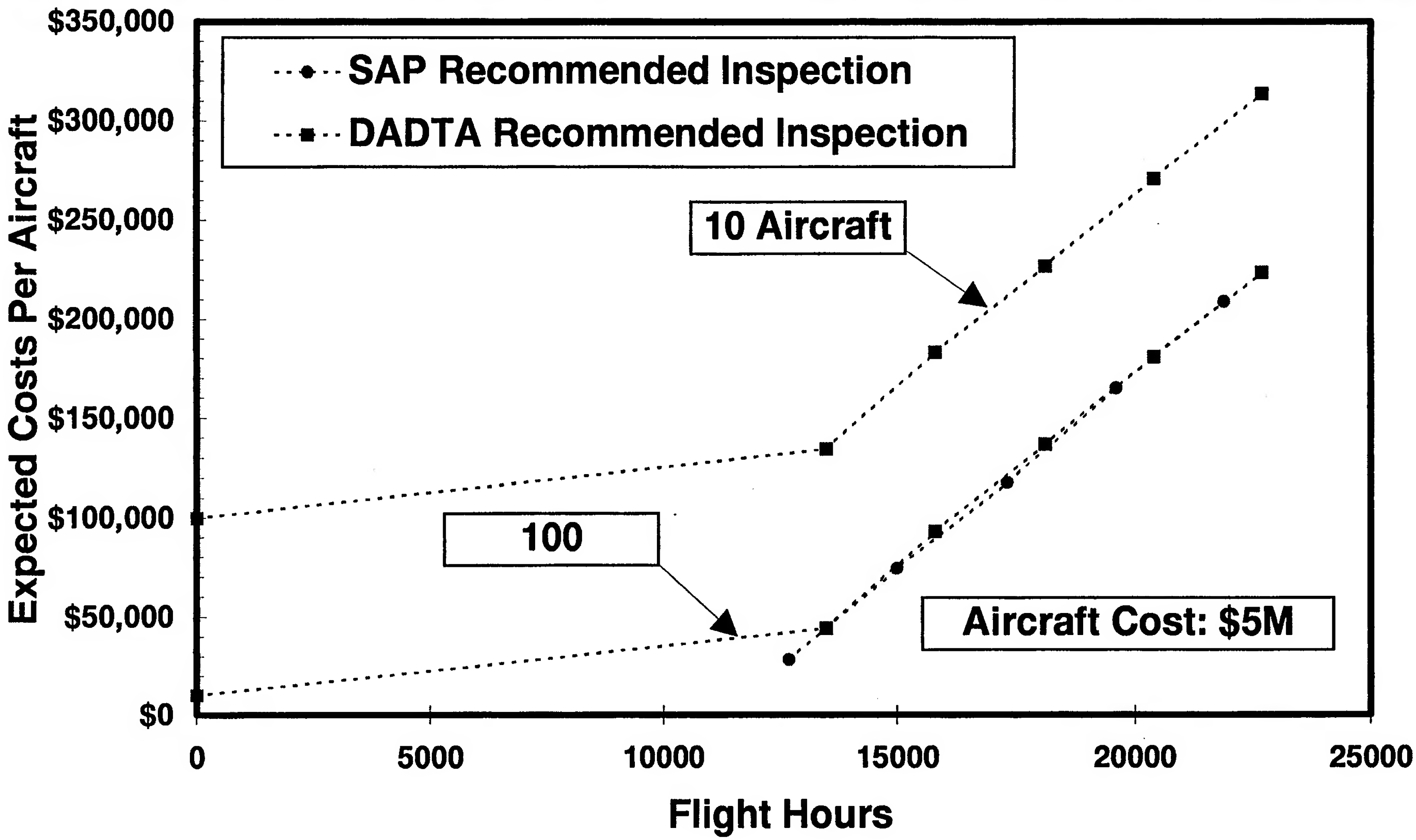
Total Costs (Including DADTA) - Usage A



Total Costs (Including DADTA) - Usage A



Total Costs (Including DADTA) - Usage C



DADTA for Military Aircraft Operators of Non-USAF Fleets

1997 USAF ASIP Conference

Observations

- Inspection and repair costs are small relative to wing replacement and aircraft loss
- Inspections based on a generic DADTA (illustrated by SAP) result in:
  - higher safety-of-flight risk
  - greater costs associated with loss of aircraft
- Inspections mandated by user-specific DADTA result in:
  - lower safety-of-flight risk
  - detected flaws prior to catastrophic aircraft loss
  - money spent in wing replacement rather than in aircraft loss

Observations

- Economic benefits of a DADTA program increase as aircraft capital investments increase (including upgraded aircraft)

- Information generated by performing a user-specific DADTA is a resource for future engineering and economic analyses
  - projection of change in usage
  - evaluation of structural and avionics upgrades

Conclusions

- Choosing <u>not</u> to perform a DADTA for user-specific usage generally results in:
  - higher risk
  - more costs
- Despite the initial costs of a DADTA, even a small fleet flown aggressively can benefit economically from a DADTA performed to the specific usage

New Approach for Fatigue Life Monitoring RNLAF F-16 Fleet

1997 USAF Aircraft Structural Integrity Program Conference, 2-4 December 1997
San Antonio, Texas

D.J. Spiekhout, National Aerospace Laboratory
Major M. Lambrichs, RNLAF
The Netherlands

NLR organisation (~ 900 persons, since 1919)

Divisions (6) :

* **Structures and Materials** (~75 persons)
* Fluid Dynamics
* Flight
* Space
* Informatics
* Electronics and Instrumentation

⇓

Departments (3) :

* **Loads and Fatigue** (~25 persons)
* Structures Technology
* Laboratory Facilities

⇓

Groups (4) :

* **Load and Usage Monitoring**
* Loads and Certification
* Engines and Failure Analysis
* Fatigue and Damage Tolerance

RNLAF organisation (~ 13500 persons, since 1913)

- Airstaff (located in the Hague)
- 2 depot facilities
- 6 operational air bases
- different weapon systems

• **fighters**	**F-16** (7 squadrons)
• transports	KDC-10, C-130, Fokker F-50 / F-60
• helicopters	Apache, Cougar, Chinook, BO-105
• trainers	Pilatus PC-7

Outline of presentation

I. Development Load Monitoring F-16 RNLAF

II. Present Load Monitoring program

III. New Load Monitoring program

IV. Description new FACE system

V. Concluding remarks

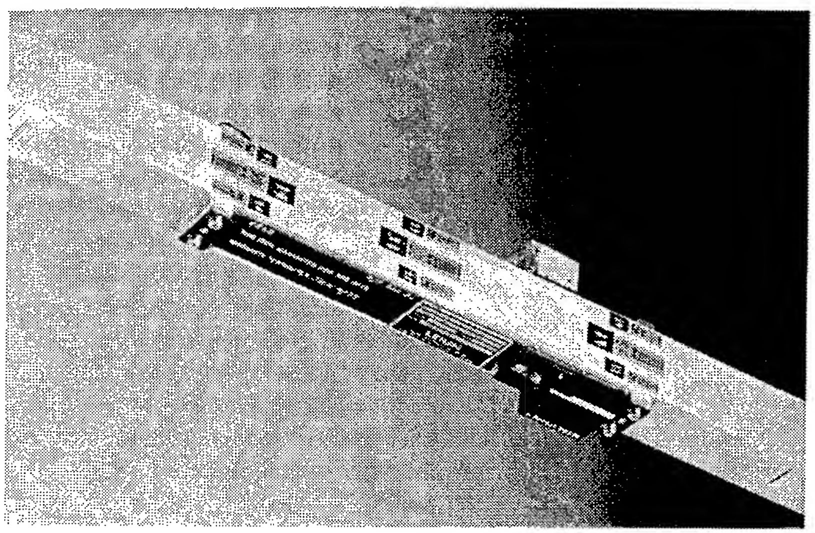
NLR

I. Development Load Monitoring F-16 RNLAF

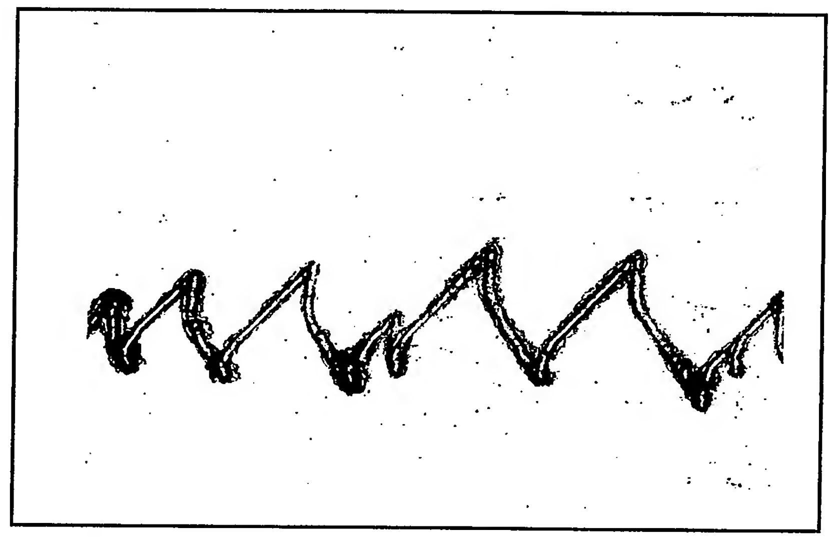
- 1979 MSR / FLR delivered with the aircraft (F-16 A/B)
- 1990 MSR replaced by 1-channel Spectrapot (Switzerland)
    - straingage at MSR location
    - on small sample of fleet
- 1992 Update to 4-channel version of Spectrapot system
    - straingage at MSR location
    - LESS type of measurements, (V, G, H)
    - engine measurements, (N2 or PLA)
    - 3 instrumented aircraft per squadron
        - more aircraft with provisions for
- 1996 Modification to new FACE system (Israel)
    - 5 straingages
    - flight parameters (MuxBus), engine (DEEC), analog
    - fleet wide

Mechanical Strain Recorder at FS 325 bulkhead for F-16 A/B load monitoring

MSR tested at NLR



Example of MSR trace



II. Present Load Monitoring program

- Inputs are measured data and CAMS data
  - measured with 4-channel Spectrapot instrumentation
  - CAMS is a logistic management system
    - mission type, store configuration, squadron, etc.

- Damage indicator is CSI (Crack Severity Index)
  - developed by NLR for F-16 monitoring
  - based on "in house" crack growth model
    - sequence effects are taken into account, (CORPUS)
  - CSI is a ratio to a reference usage (FSMP 1985)
  - also in use by LMTAS

NLR

Reports to Airstaff

- **Quarterly reports**
  - per tail number fatigue damage calculated
    - for last period and from "0" hours
  - changes in usage / loading experience reported
    - damage hours , (CSI * flight hours)
    - mission mixture
    - store configuration usage
- summary reports to squadrons by airstaff
- **Half yearly reports**
  - global statistics of measuring program
    - CSI overviews
    - comparison "sample" size with "all" flights
- **Ad hoc reports**
  - on special request airstaff

NLR

III. New Load Monitoring program

Development stages

- airstaff is interested in fleet wide monitoring
  - more switching of aircraft between squadrons
- after “Falcon Up” modification need for more strain gages
  - mid fuselage section expected to give less problems
  - more problems in other areas of aircraft experienced

- 1993 test flight with “ACE” system in the Netherlands
  - NLR specified FACE system for load monitoring
- 1994 test flight with redesigned system, (proof of concept)
- 1995 contract signed
- 1996 first test flights with prototype new FACE system
  - first production systems delivered
- 1997 approved configuration for MLU by LMTAS

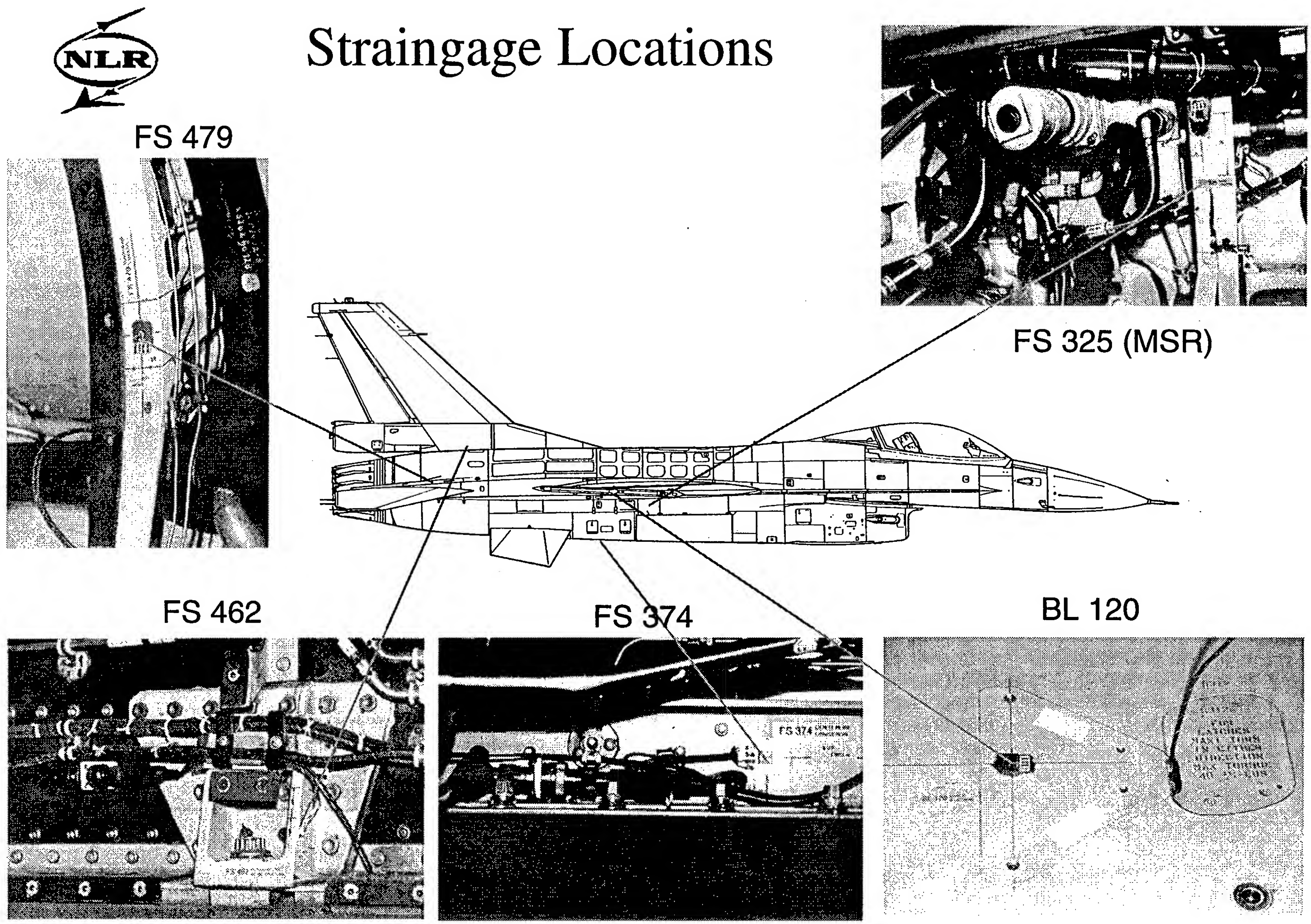
Main features of program

- Program set up as a <u>“sample program”</u>
  - 100 % data capture is an illusion
  - exchange rate of cartridges is 25 flight hours
  - RNLAF wants fast feedback to squadrons
- Weekly update from CAMS (RNLAF) data base
  - usage per tailnumber (mission mixture, hours, etc)
  - modification status of each aircraft
- Weekly CSI data per aircraft / squadron calculated
  - update of 5 CSI values for each tailnumber
  - calculated from <u>statistical CSI and CAMS data</u>

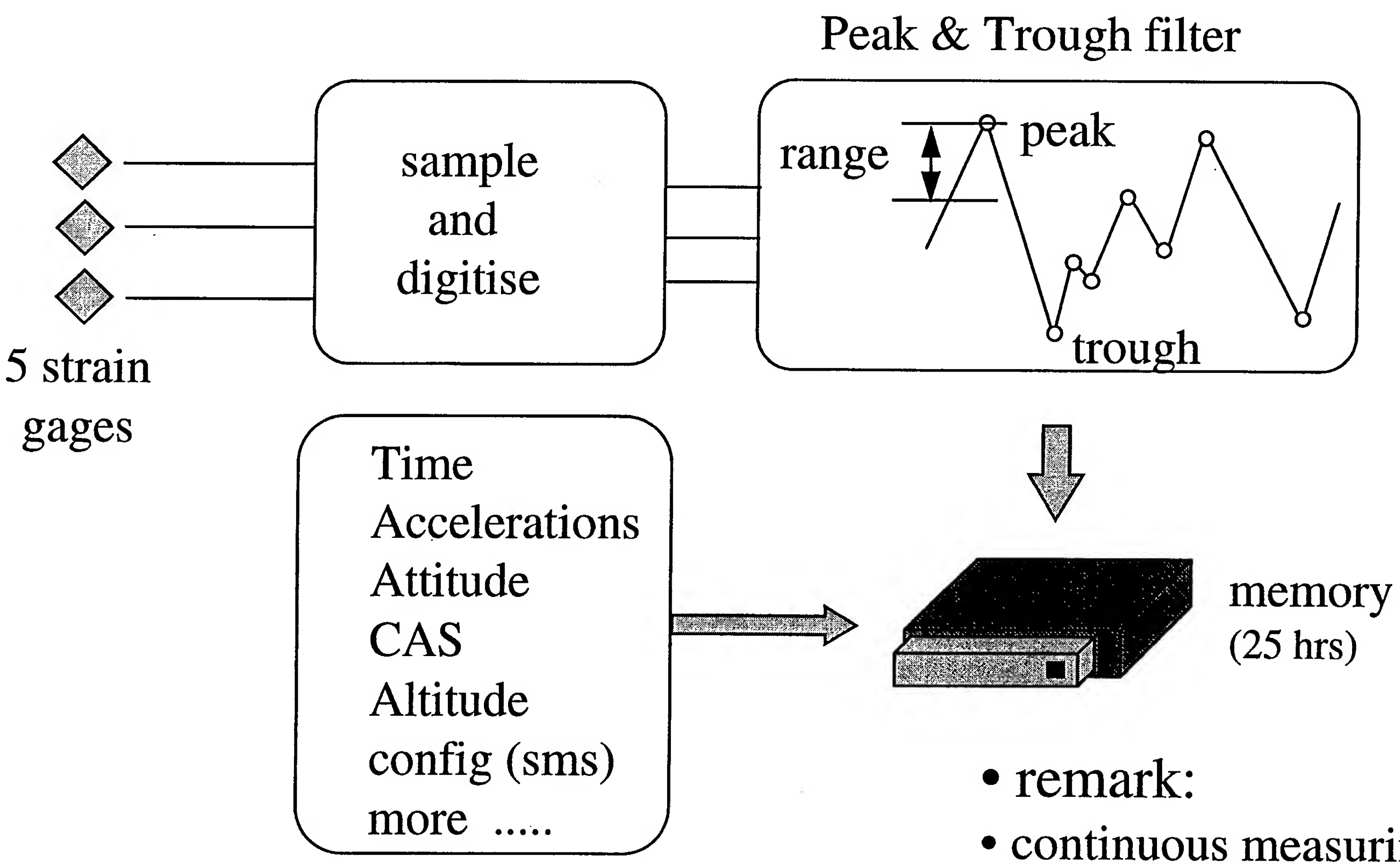
- CSI damage calculation
  - reference strain sequences calculated by LMTAS
  - as a reference the 1985 usage has been taken (FSMP)
  - more than 31 ASIP points will be monitored, (MLU)
  - measured data will be stored in a data base
    - replace statistical data for particular aircraft
    - upgrade statistical data per squadron

Straingage instrumentation

- 5 straingage bridge locations selected
  - centre section fuselage / inner wing
  - aft fuselage
  - outer wing
  - tail sections
- Selection based on
  - discussions with LMTAS
  - special measuring program
    - two instrumented aircraft
    - 10 straingage bridges
    - operational flights

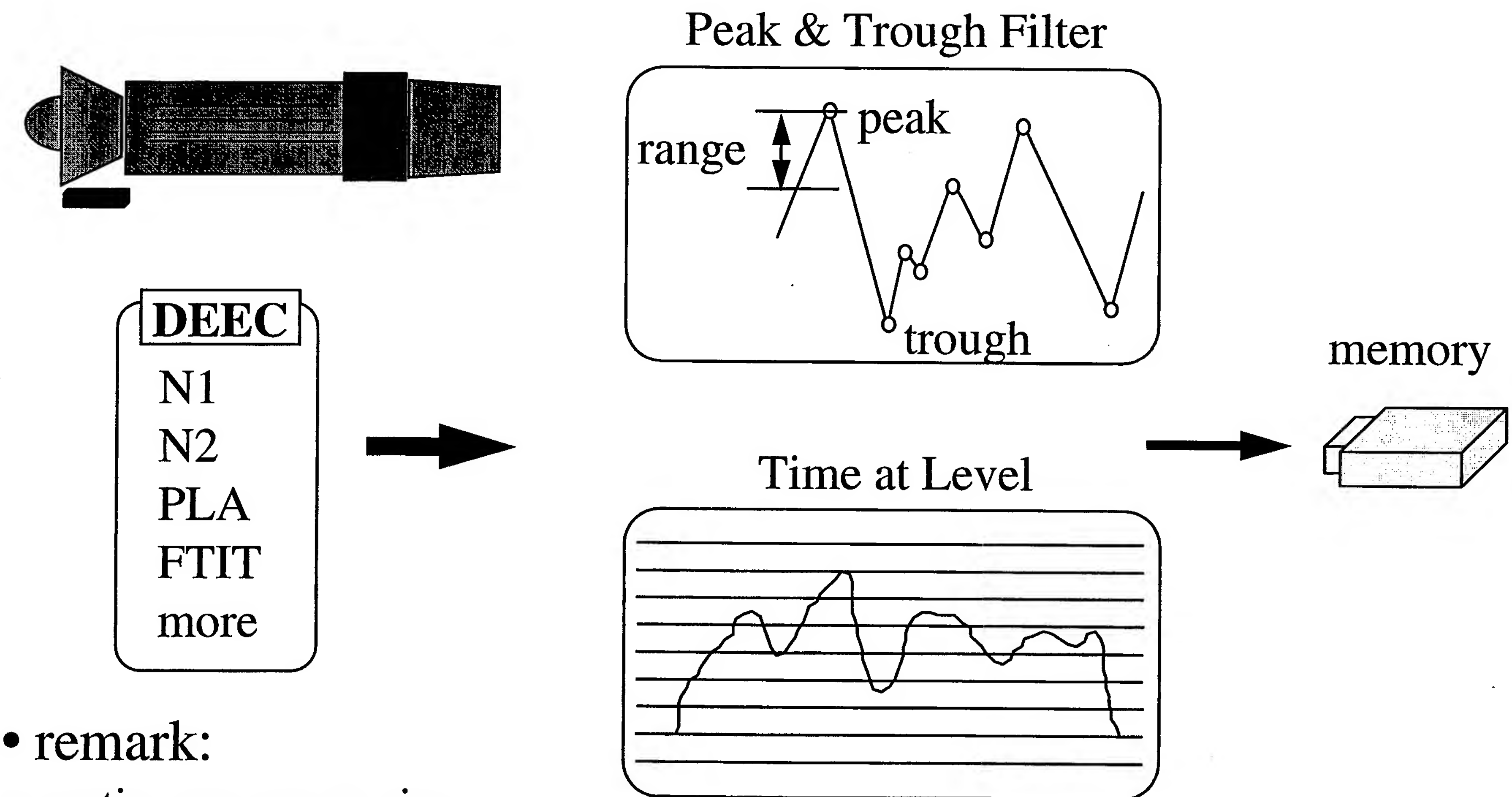

Straingage Locations
NLR
FS 479
FS 325 (MSR)
FS 462
FS 374
BL 120

Airframe Fatigue Monitoring



- remark:
- continuous measuring
- on board data reduction

Engine Fatigue Monitoring



- remark:
- continuous measuring
- on board data reduction

Data storage /Analysing and Reporting

- ORACLE <u>relational data base program</u> has been chosen
  - all calculations will be done in this program
  - reports will be generated by this program
- On line results available for airstaff and squadron
- "Automatic" data storage from squadron to NLR

IV. Description new FACE system

- Tasks
  - airframe load monitoring
  - engine load monitoring
  - avionics health monitoring
  - ACE (air combat evaluation)
  - mishap investigation tool (VADR)
- Covering 3 different aircraft configurations
  - Avionics (MLU / OCU), Engine (EEC / DEEC)
- Ad hoc measurements can be easily added
  - software very flexible for initiating “next flight”
  - by means of “set up configuration” input file

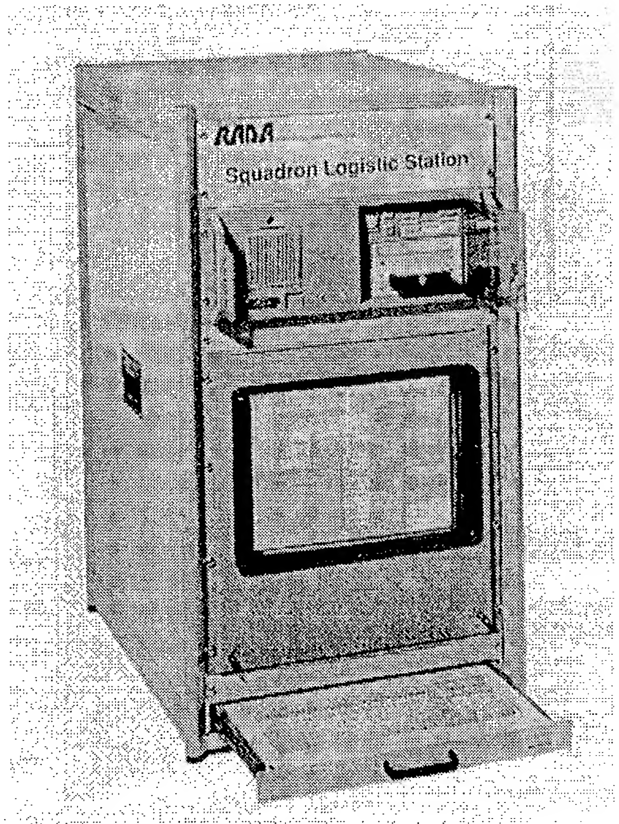
Flexible software for initiating "next" flight"

- Master / slave concept chosen
  - up to 15 masters
  - up to 50 slaves per master
- Sample rate can be chosen
  - analog up to 1000 Hz
  - MuxBus up to 50 Hz
  - DEEC up to 5 Hz
- Data reduction
  - peak and trough (PAT)
  - time at level (TAL)
  - periodic (SAMPLE)
- Filter frequency can be set
- Selective recording (flight mode, triggers)

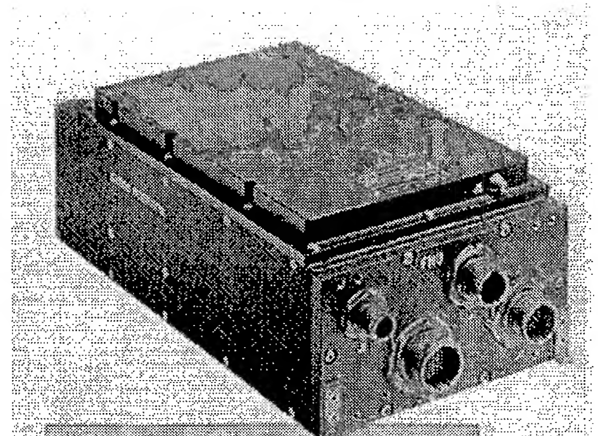
⟹ Result: Set Up Configuration input file (SCF)

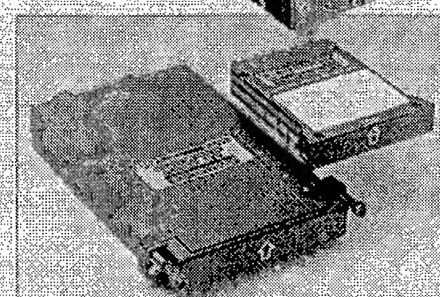
Ground and airborne equipment

LDS



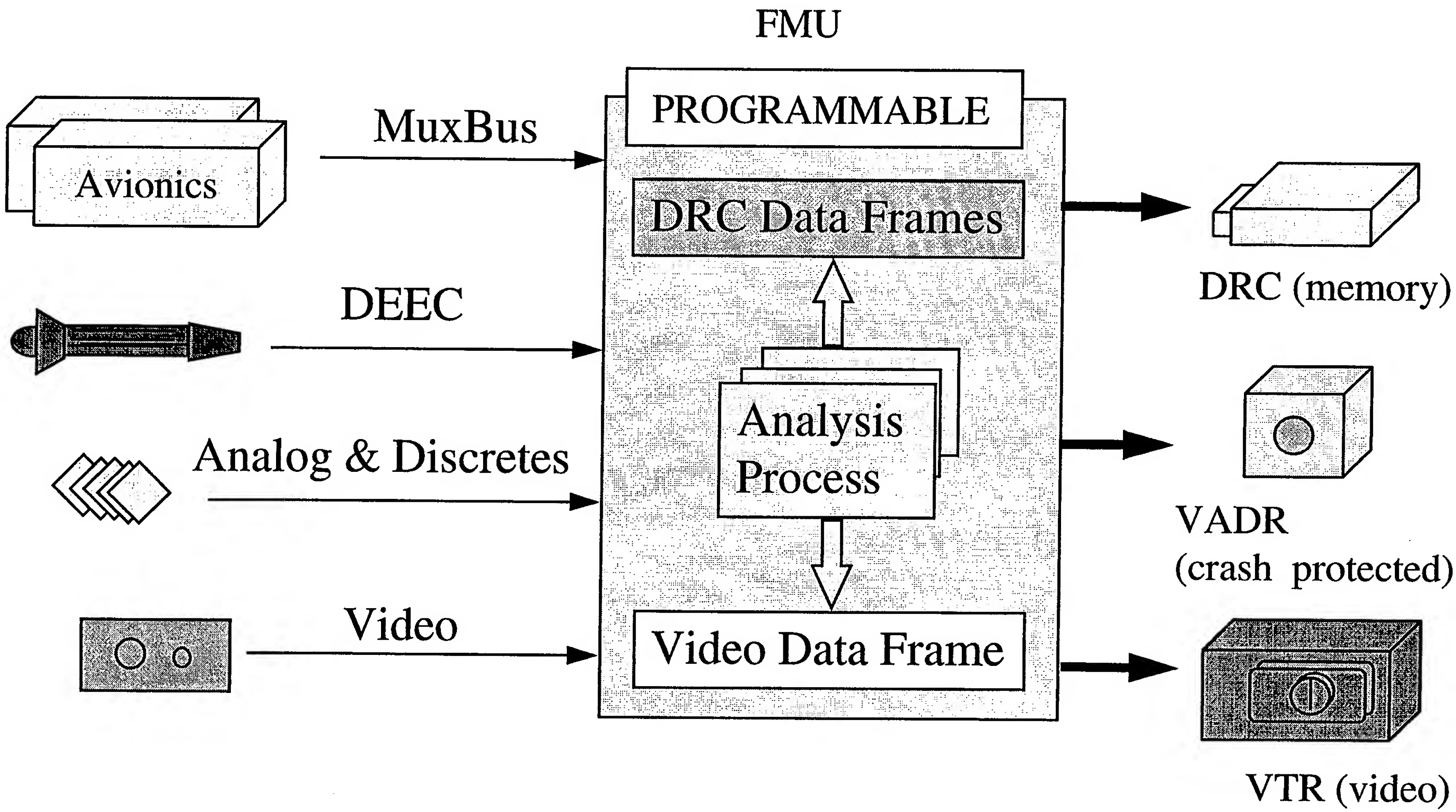
FMU



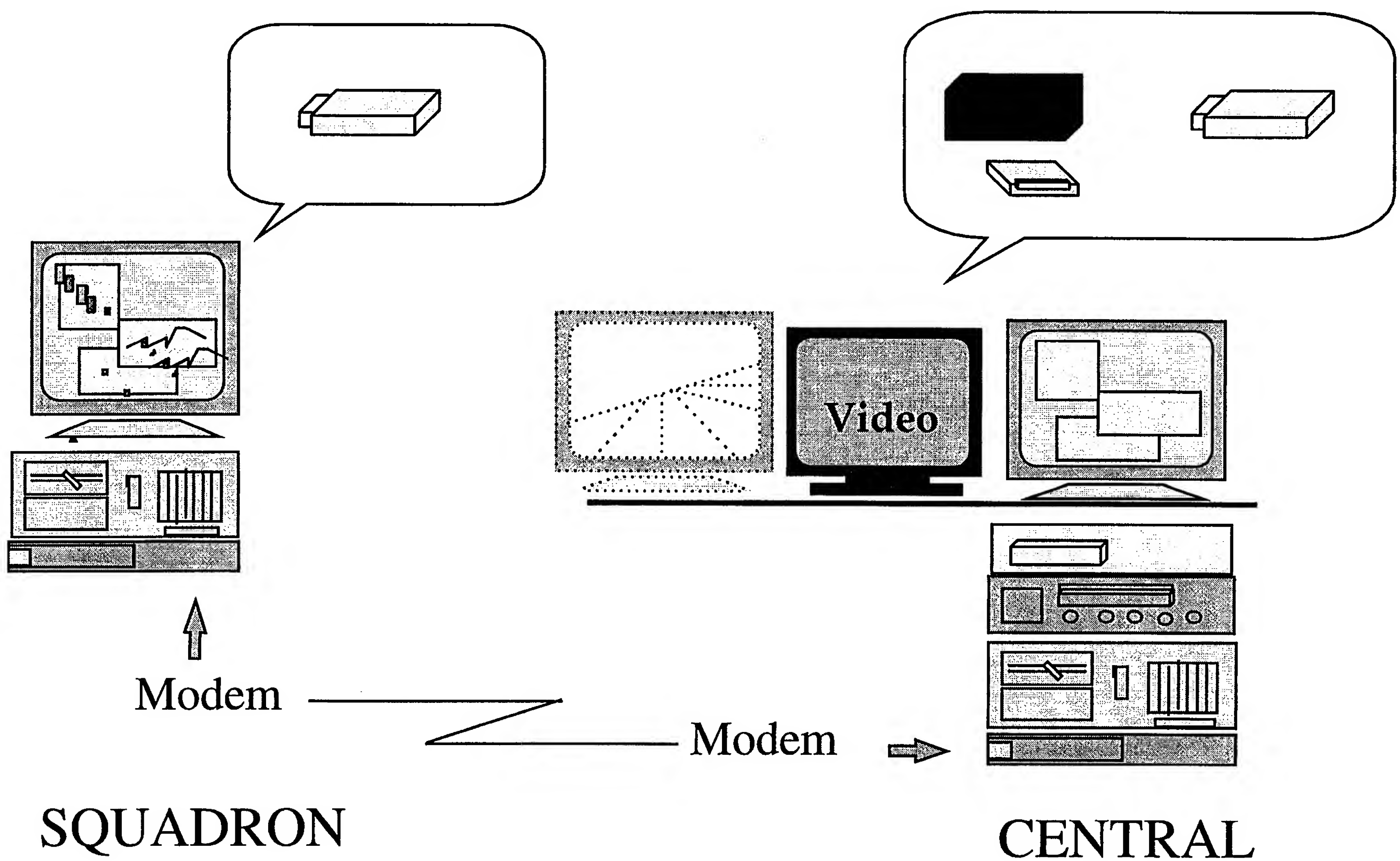


DRU/DRC

Overview FACE system structure



Ground Logistic Debriefing Stations

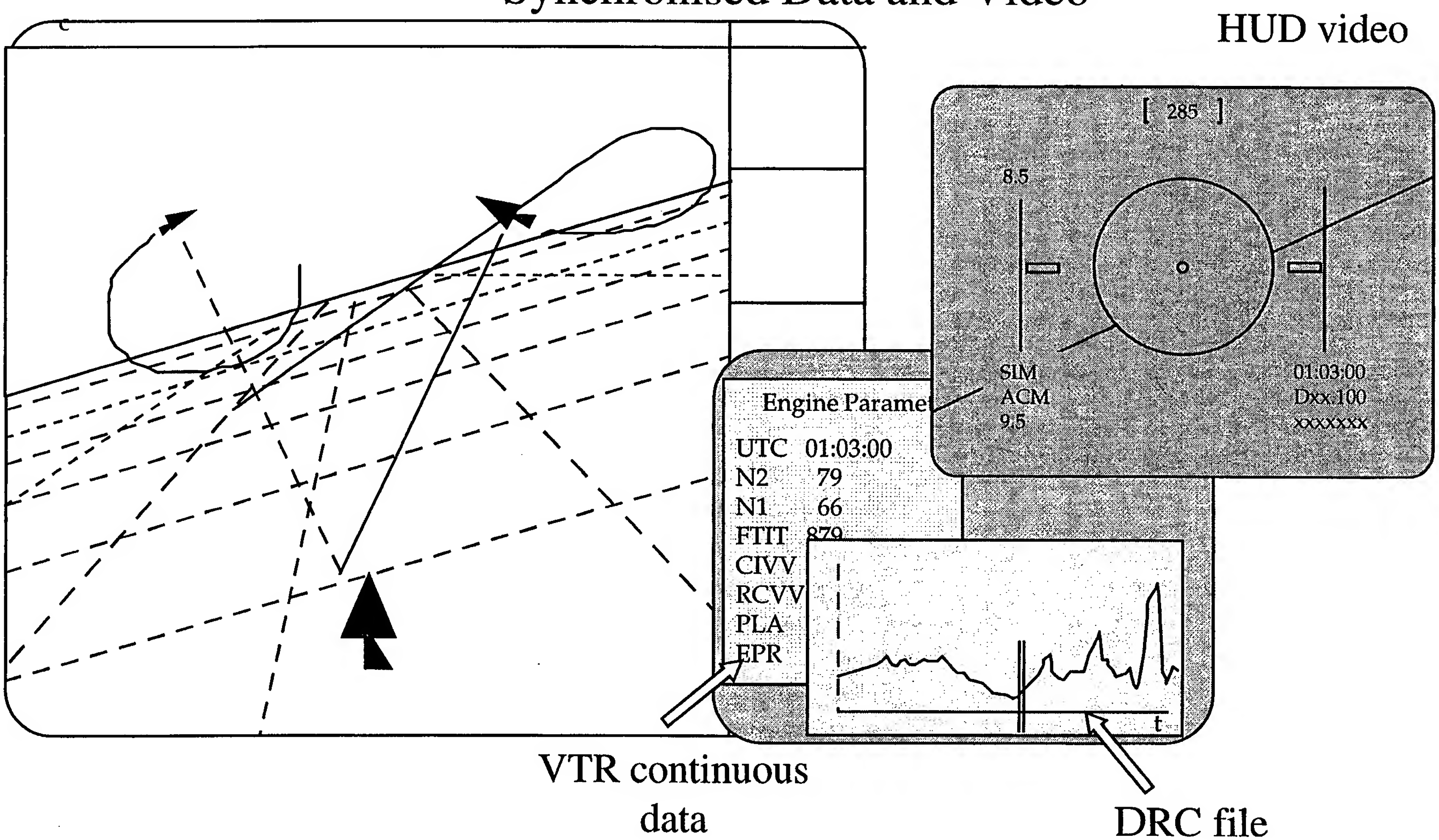


Main tasks (Central) Logistic Debriefing Station

- FACE software and SCF management
- Analysis structural fatigue (CSI calculation)
- Analysis engine fatigue (TAC calculation)
- Accident / incident investigation
  - from VCR, DRC & VADR files
- Avionics & engine malfunctions trouble shooting
  - MLF and PLF reports, circular buffer
- Analysis of flight manoeuvres
- Ad hoc investigations
  - risk assessment

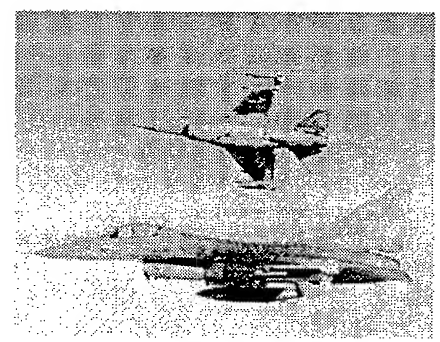
Typical Central Station Displays

Synchronised Data and Video



V. Concluding remarks

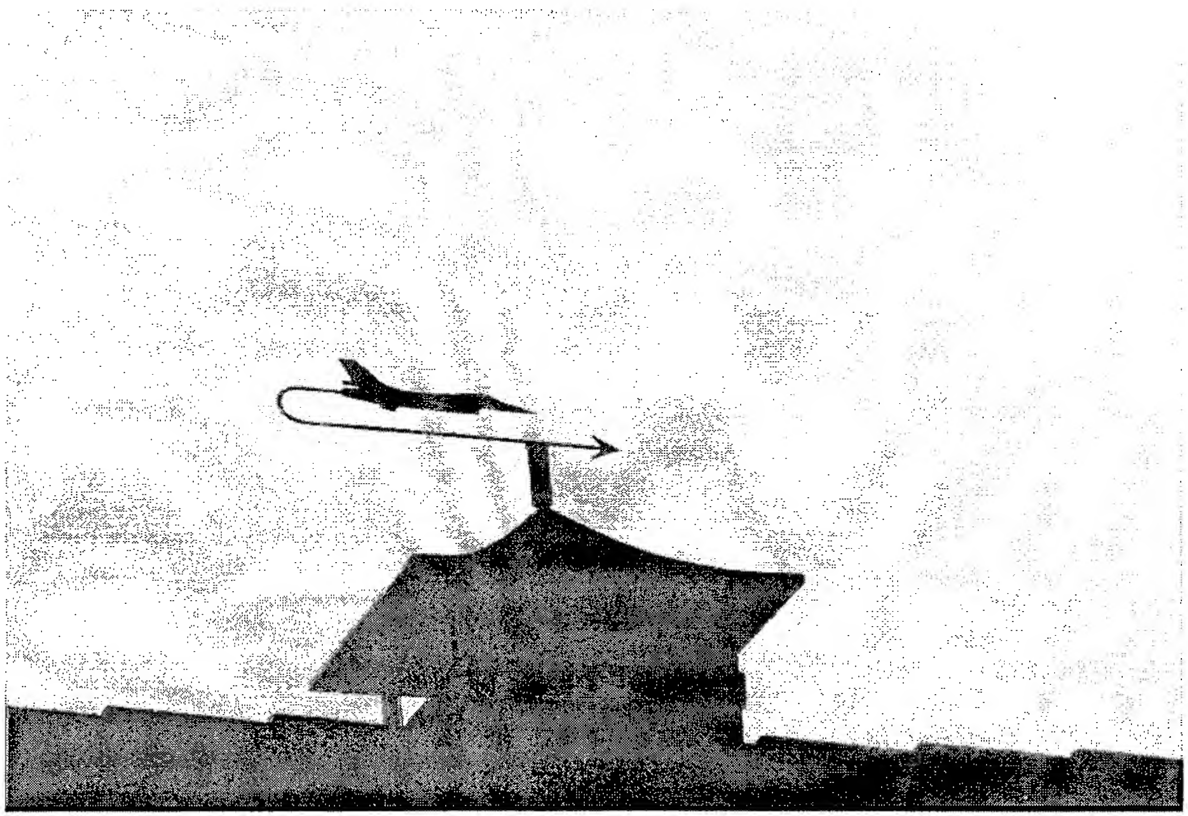
- Very flexible instrumentation tool available
  - package fully certified by LMTAS for MLU F-16
- Load monitoring based on more straingages
  - detailed advice to pilots about operational usage
- RNLAF modified about 50 aircraft sofar



"Train as you fight . . . .

. . . . Maintain as you train"

A bright future lies ahead for RNLAF /FACE



XF-2 (JAPANESE NEXT-GENERATION SUPPORT FIGHTER) FULL SCALE STATIC TESTS AND DURABILITY TEST - INTERIM TEST RESULTS

Masami KAGEYAMA, Noboru ARATA,
Akira YOKOYAMA, Hiroshi TAGUCHI,
Akira KUBO, Yasuo OTANI,
Kofu OKUDA, Makoto UEMURA
and Yasuhiro ISAKARI (JDA)

XF-2 Full Scale Static Strength Tests

Purpose

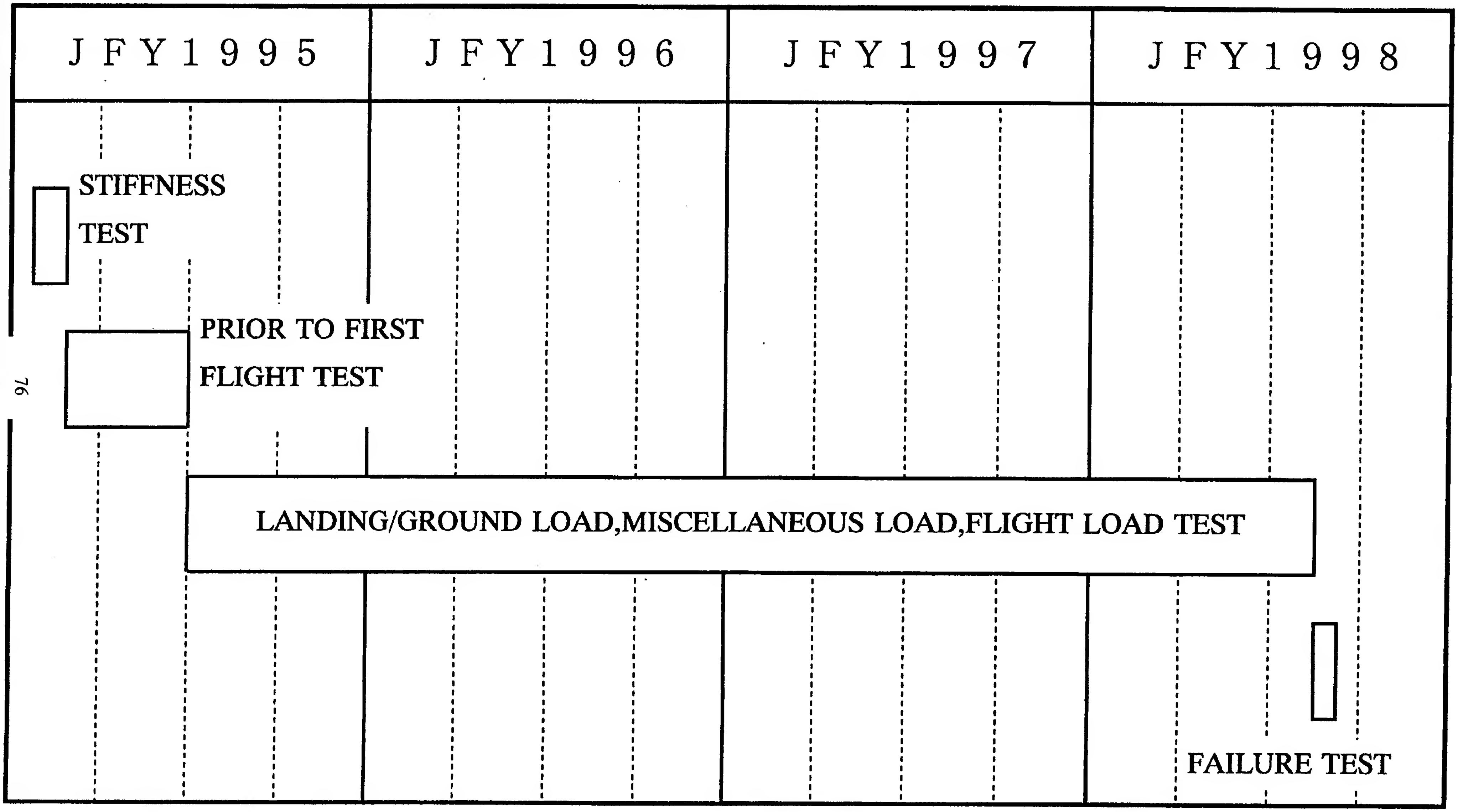
(1) To certify that the test article has enough static strength for design loads

(2) To demonstrate that LEF and TEF perform satisfactorily with applicable maximum operating loads

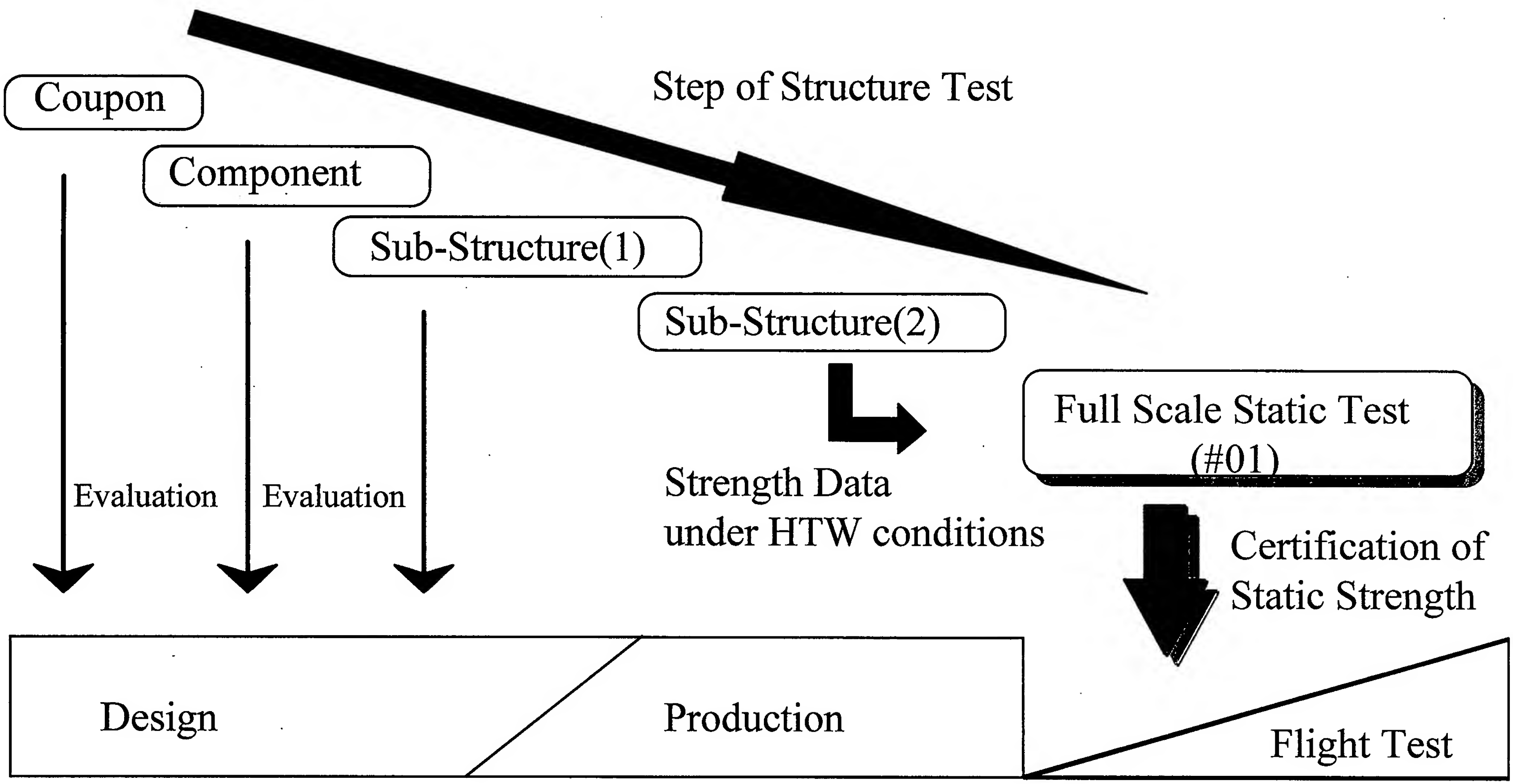
(3) To obtain the test article's stiffness data

(4) To obtain overall airframe static strength and failure mode

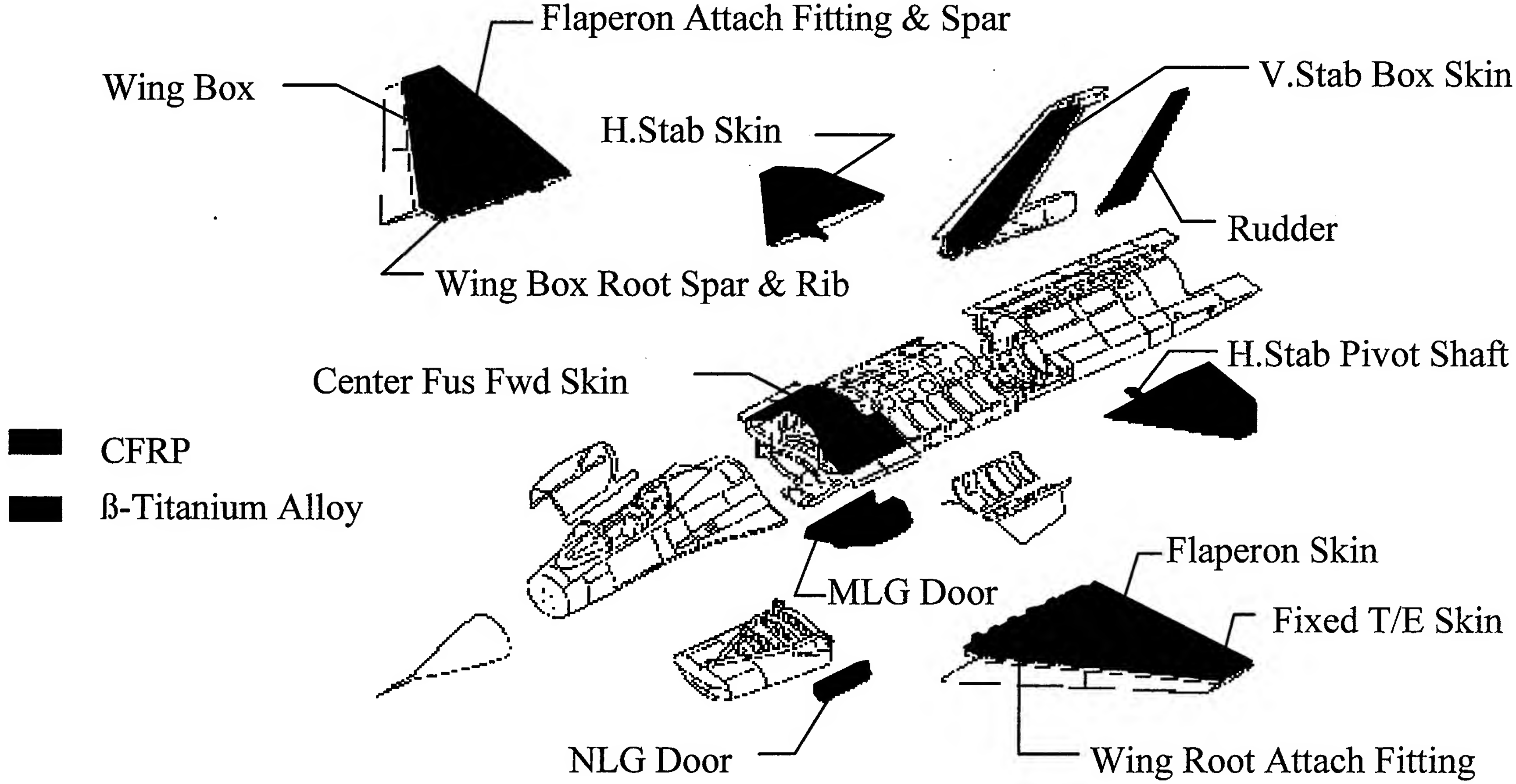
FULL SCALE STATIC TEST PLAN



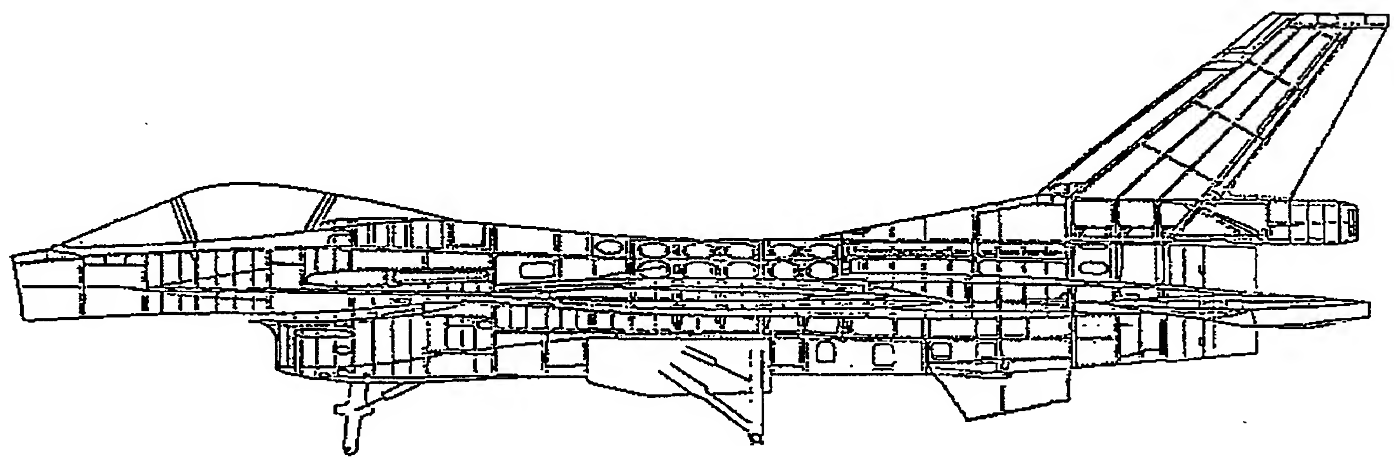
Position of #01 Test Status



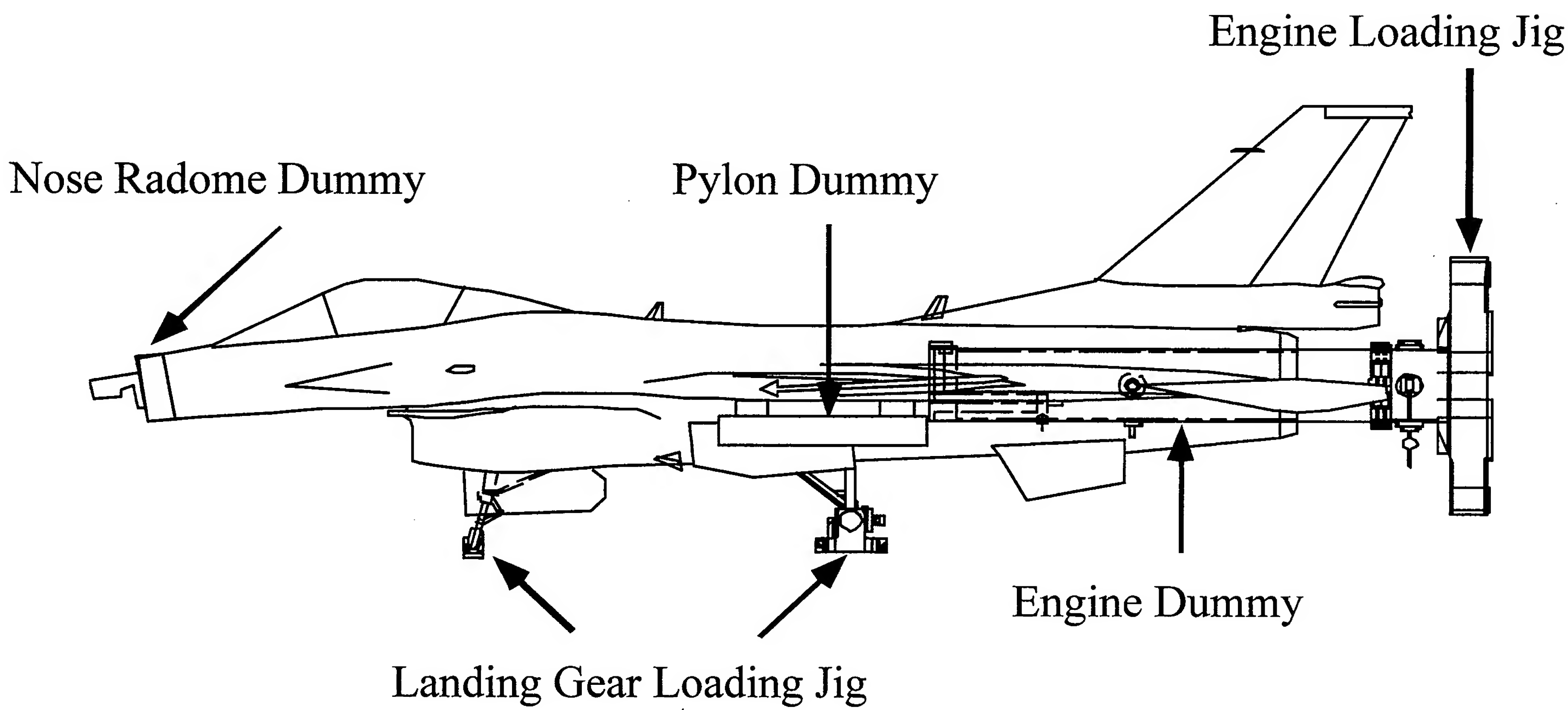
XF-2 Material Application



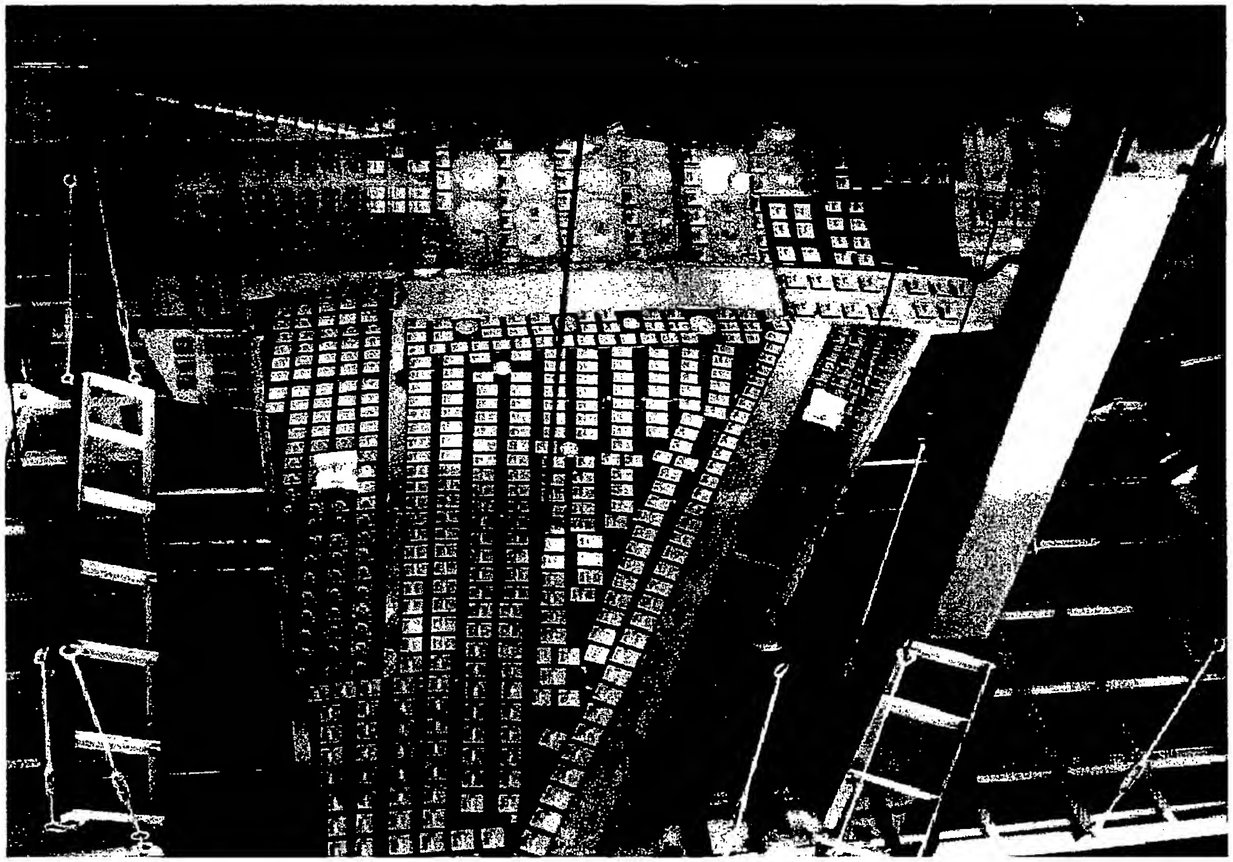
#01 Test Article



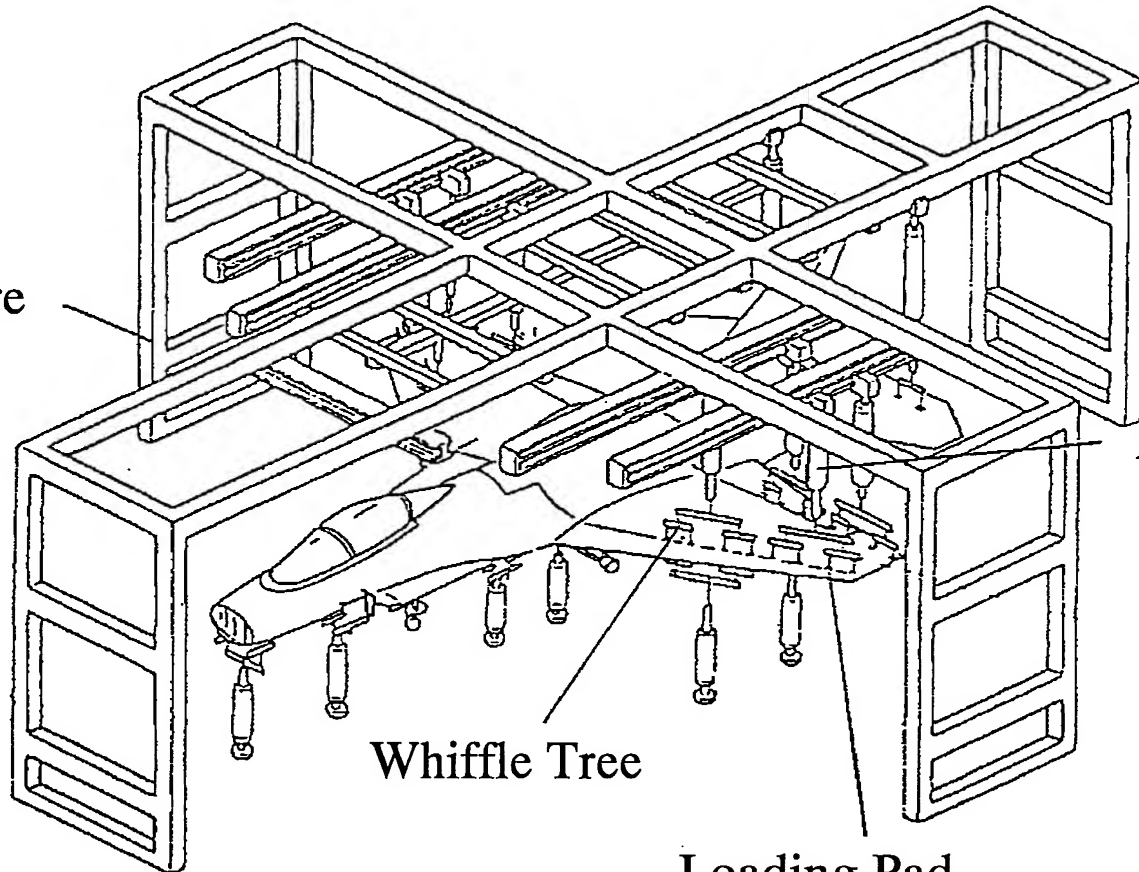
Loading Jig / Dummy



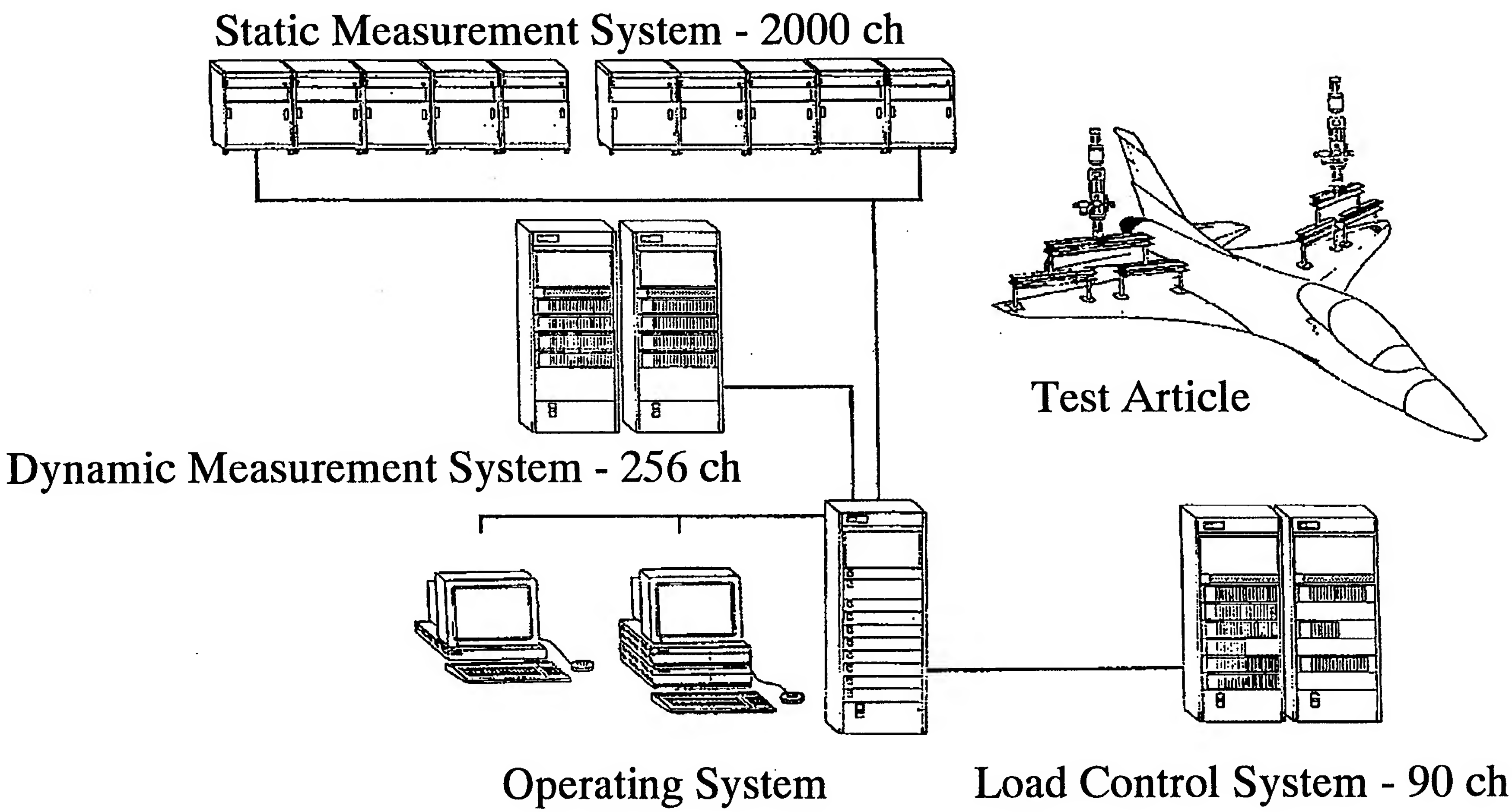
Set Up View



Loading System



Test Systems



Policy of Static Strength Certification

(1) To certify that the main airframe of XF-2 has satisfactory static strength, static tests are conducted for design load cases under room temperature dry(RTD) conditions.

(2) To certify the composite structure, RTD test results are supplemented using hot temperature wet(HTW) strength data.

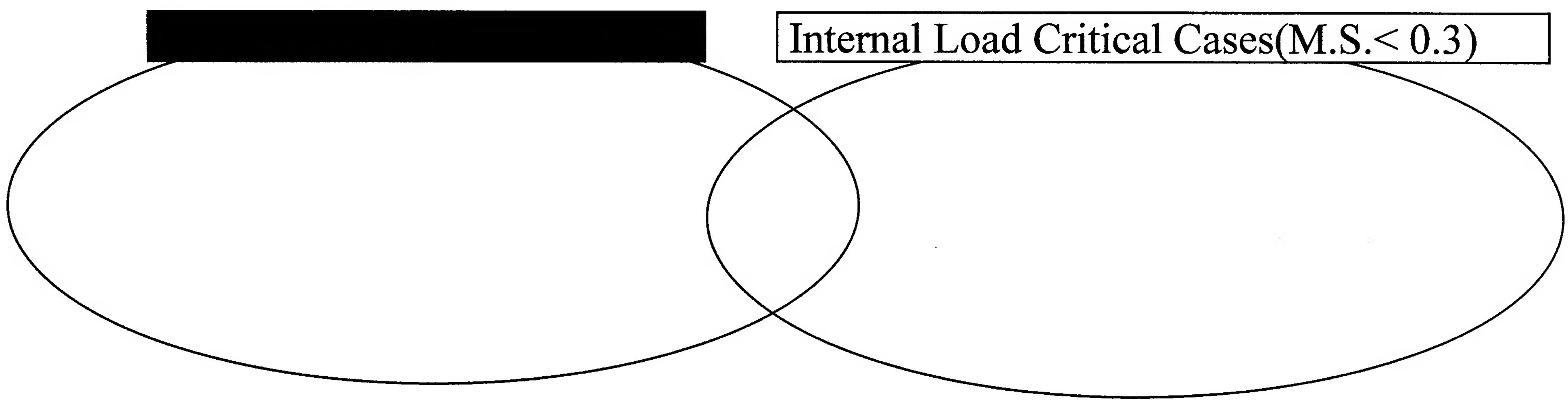
Selection of Test Cases

All Design Load Conditions

↓

Load Conditions for Flight Safety

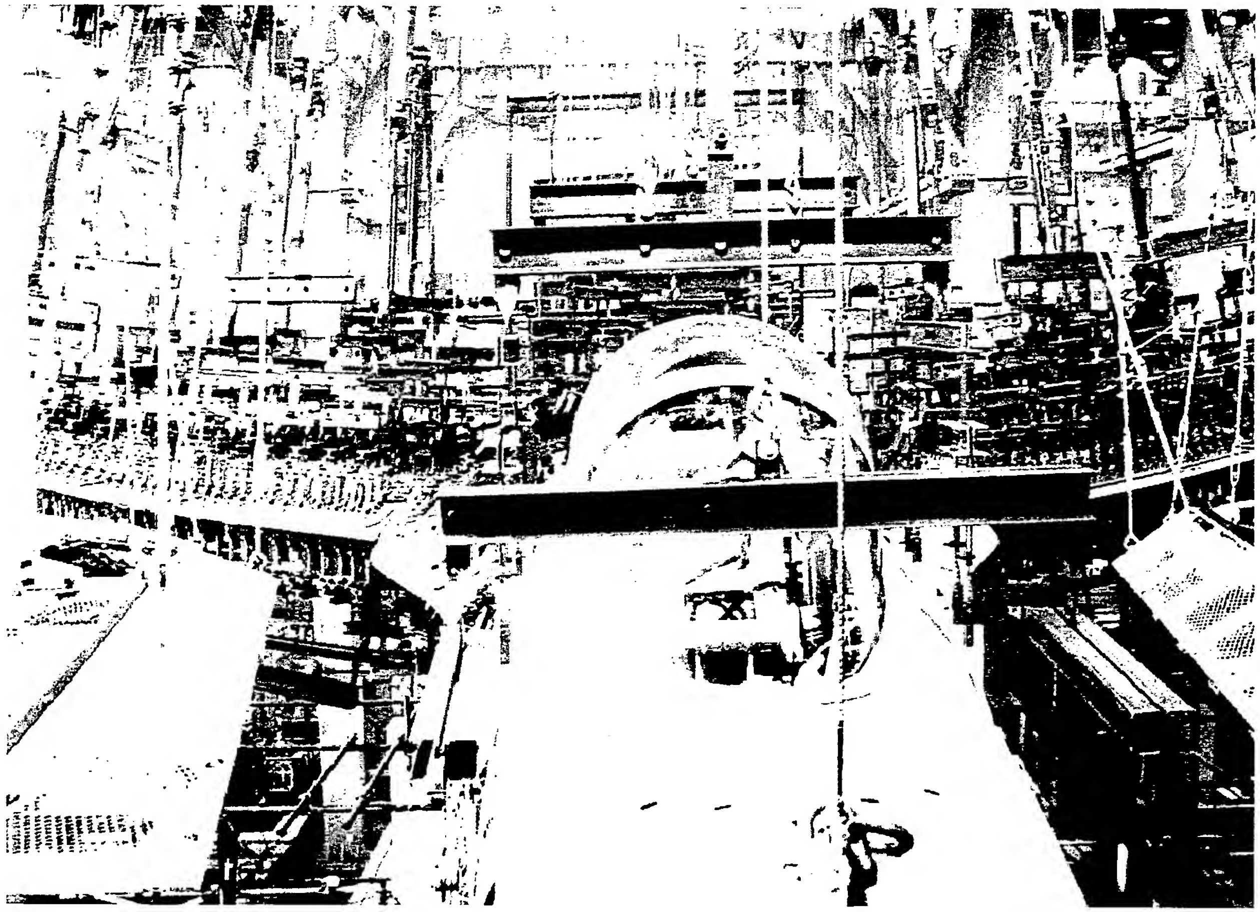
↓



Test Cases

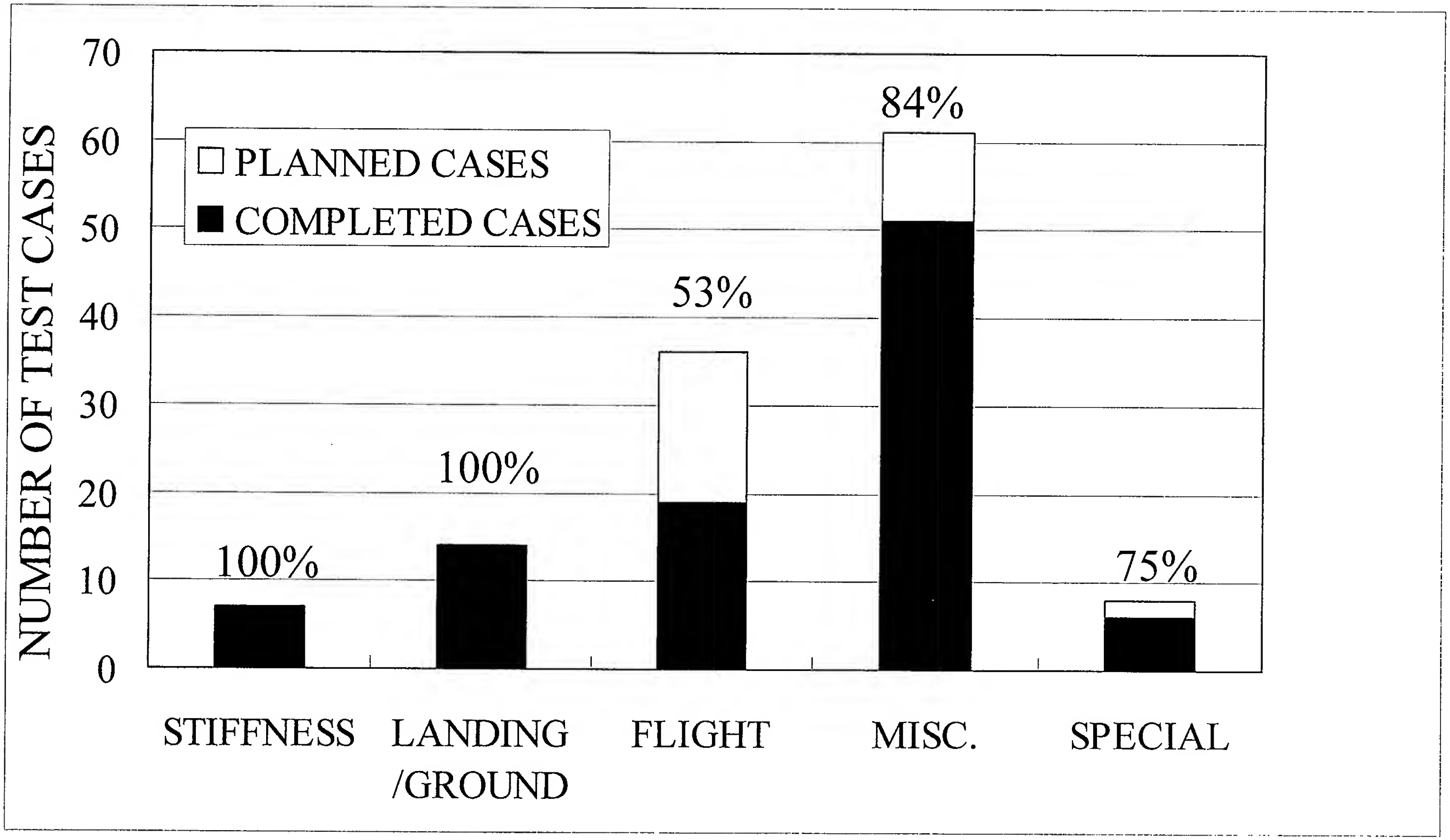
Category	Test Name	Number of Test Cases
Stiffness	Wing, Fuselage, etc.	7
Landing & Ground Load	NLG Spin-up, Spring-back, MLG Spin-up, etc.	14
Flight Load	SPU, RPO, ROLL, etc.	36
Misc. Load	External Store, Drag Chute, Speed Brake, etc.	61
Special Load	Cockpit Pressure, Failure, etc.	8
Total		126

Test View



Symmetrical Pull Up - 100% Design Load

Test Status



NUMBER OF TEST CASES
70
60
50
40
30
20
10
0
PLANNED CASES
COMPLETED CASES
100%
100%
53%
84%
75%
STIFFNESS
LANDING /GROUND
FLIGHT
MISC.
SPECIAL

Conclusion

(1) Prior to 1st flight (Oct. 1995), all 28 static test cases required for 1st flight and company flight certification were successfully completed.

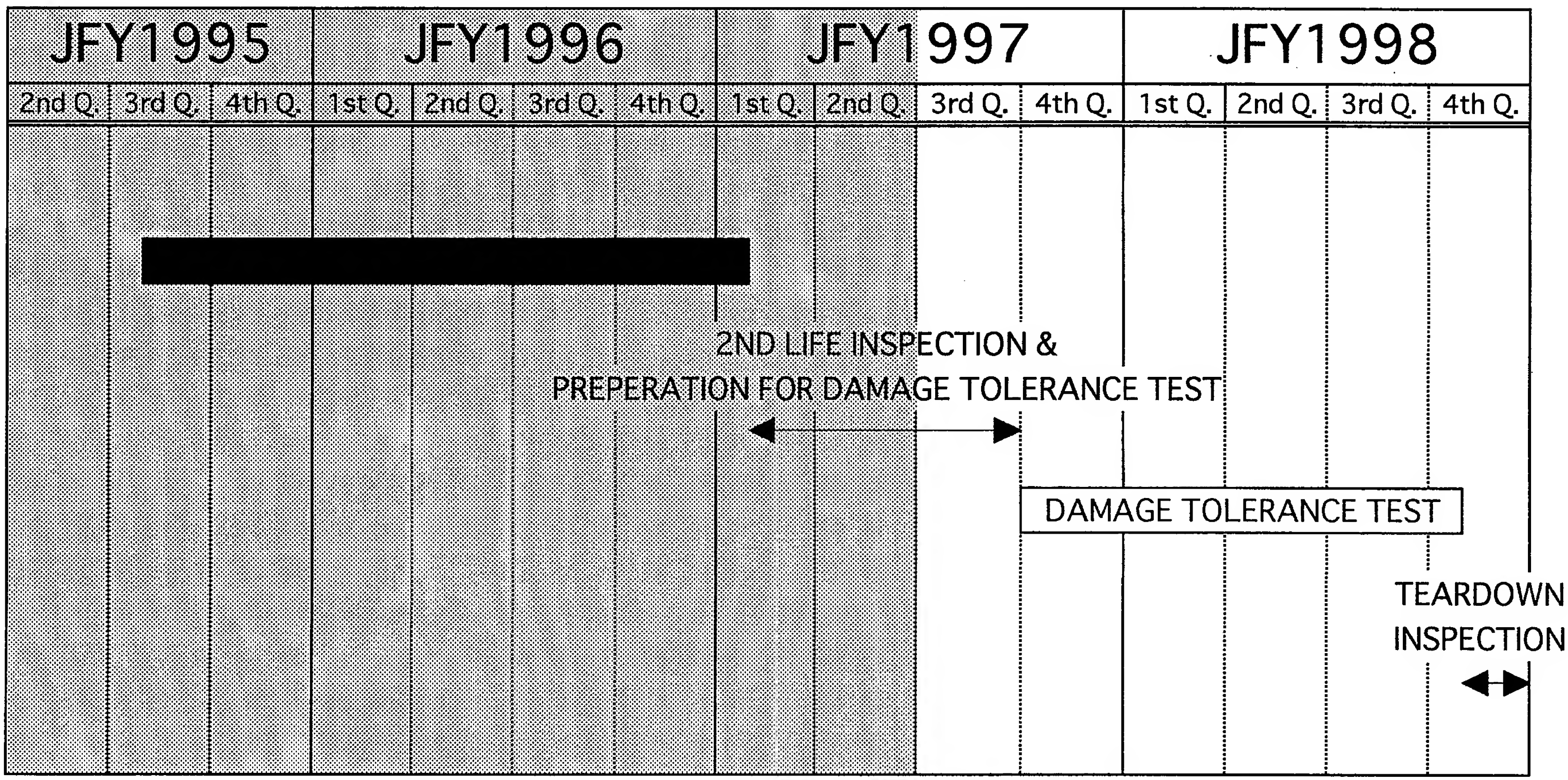
(2) As of Dec. 1997, 97 out of 126 required static test cases have been successfully completed. While analyzing and evaluating test results, we are conducting the remaining test cases.

XF-2 Full Scale
Fatigue Strength Tests

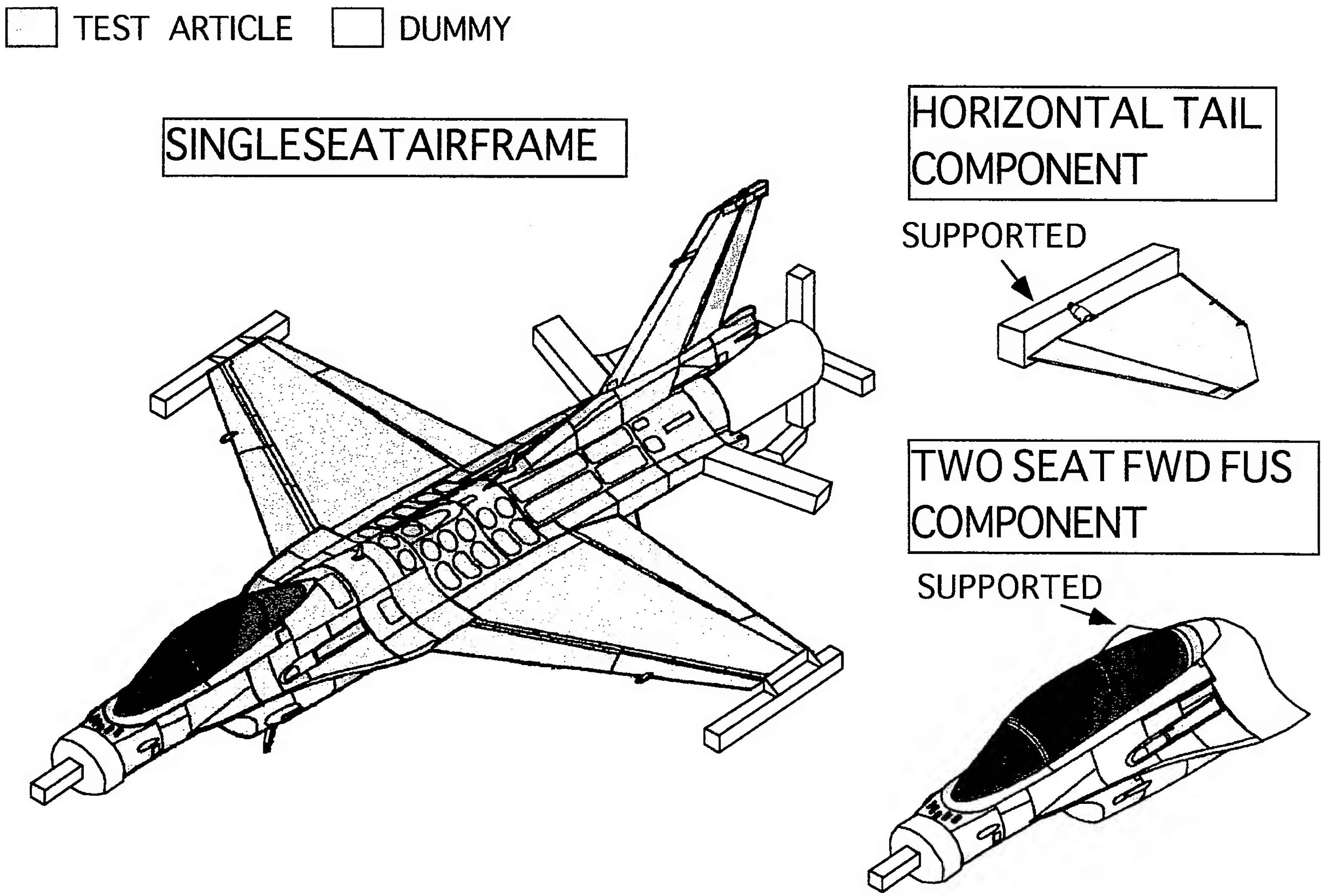
PURPOSE

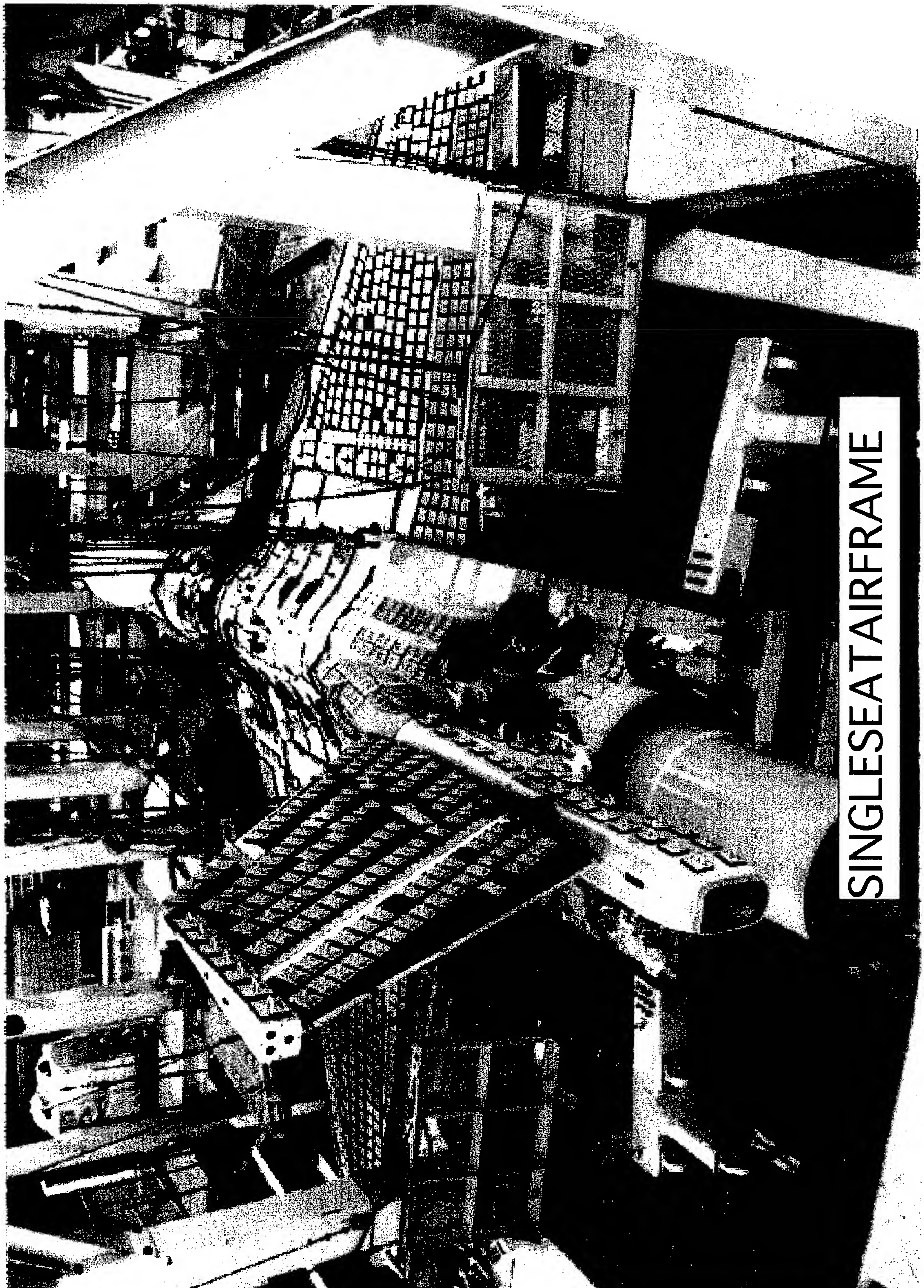
1. CERTIFICATION OF THE DURABILITY UNDER THE DESIGN LOAD SPECTRUM
2. CERTIFICATION OF THE HOT SPOTS
3. OBTAINING DATA FOR FORCE MAINTENANCE OR MODIFICATION
4. CERTIFICATION OF THE DAMAGE TOLERANCE

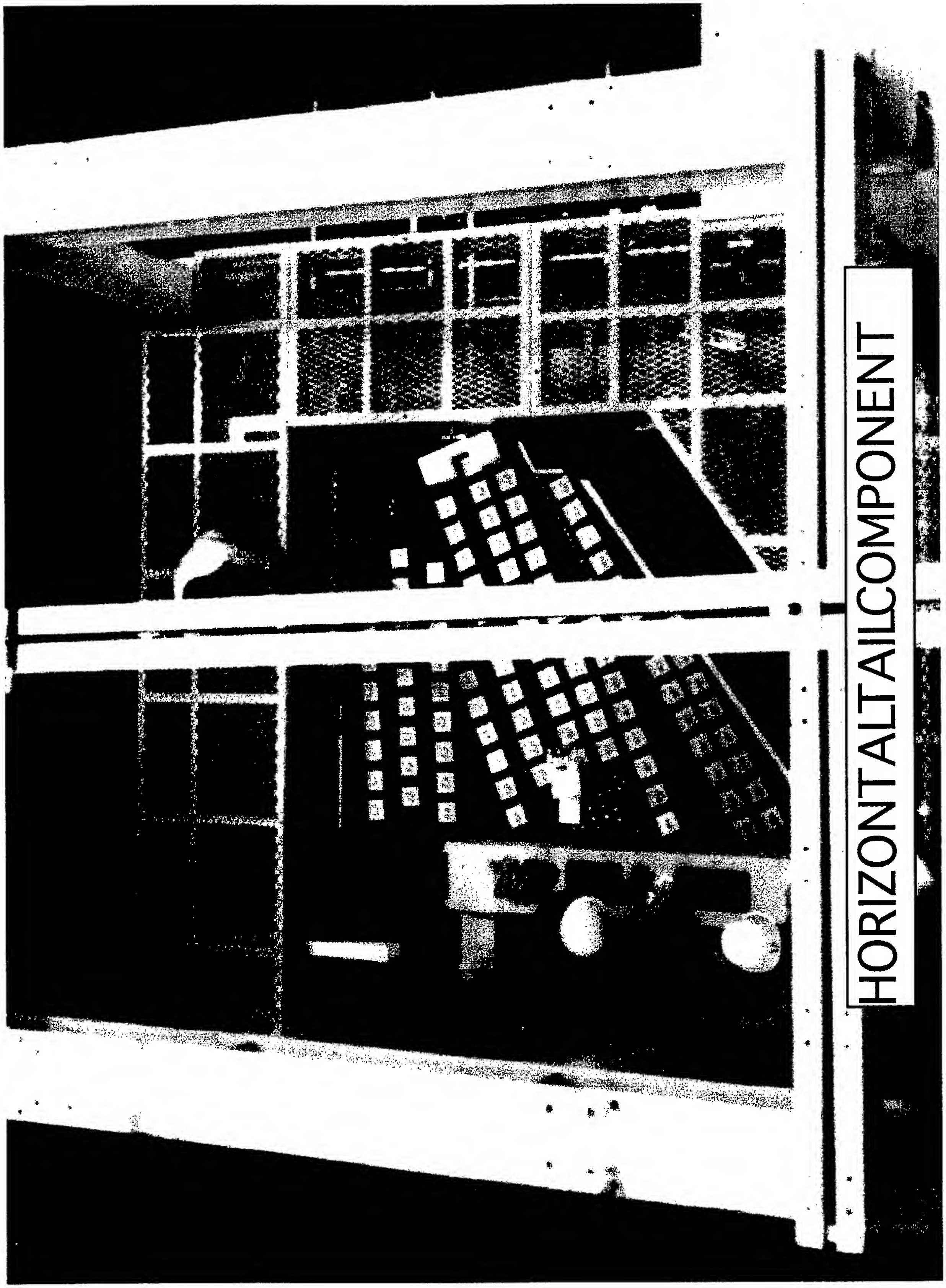
FULL SCALE FATIGUE TEST SCHEDULE

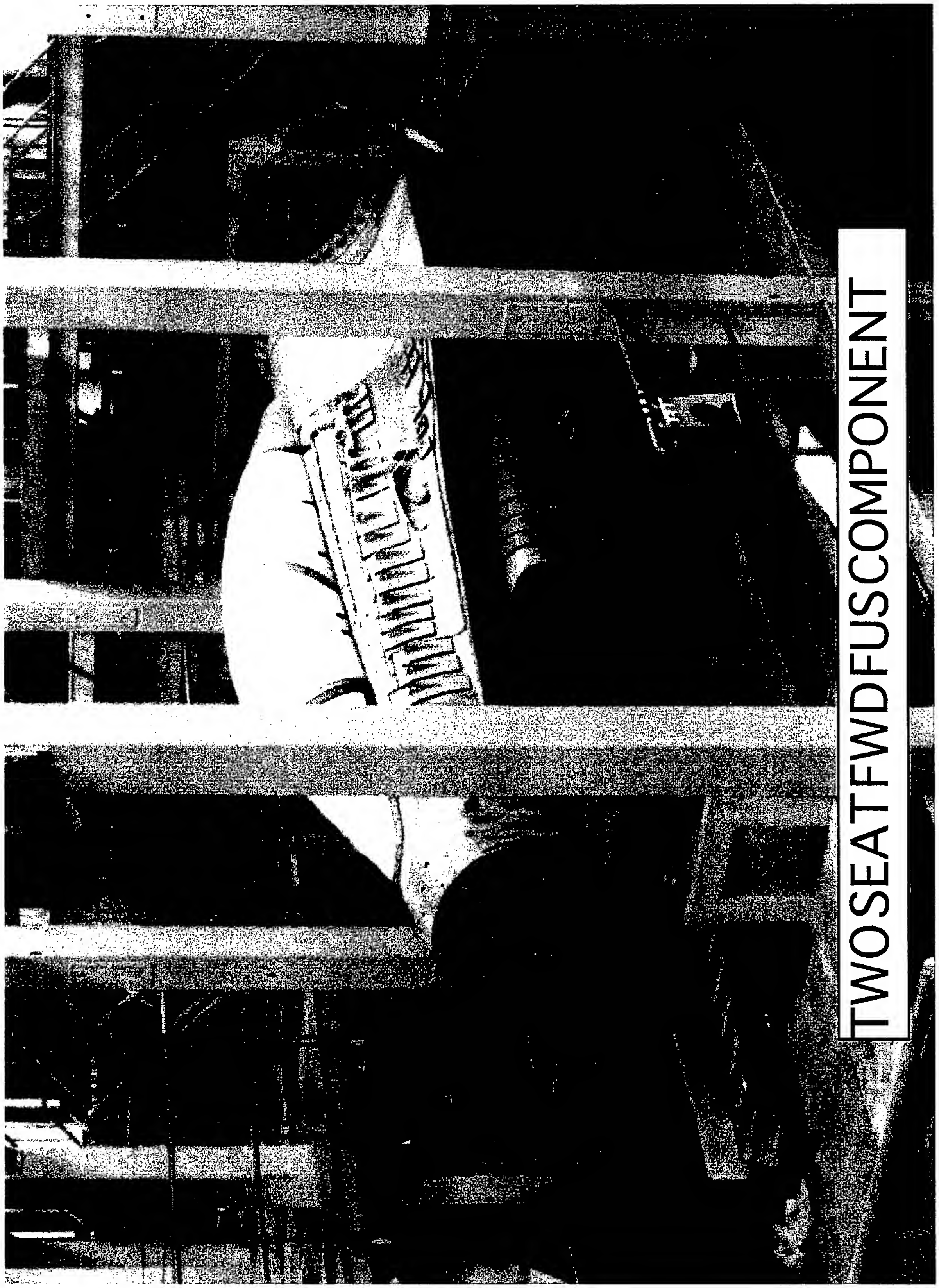


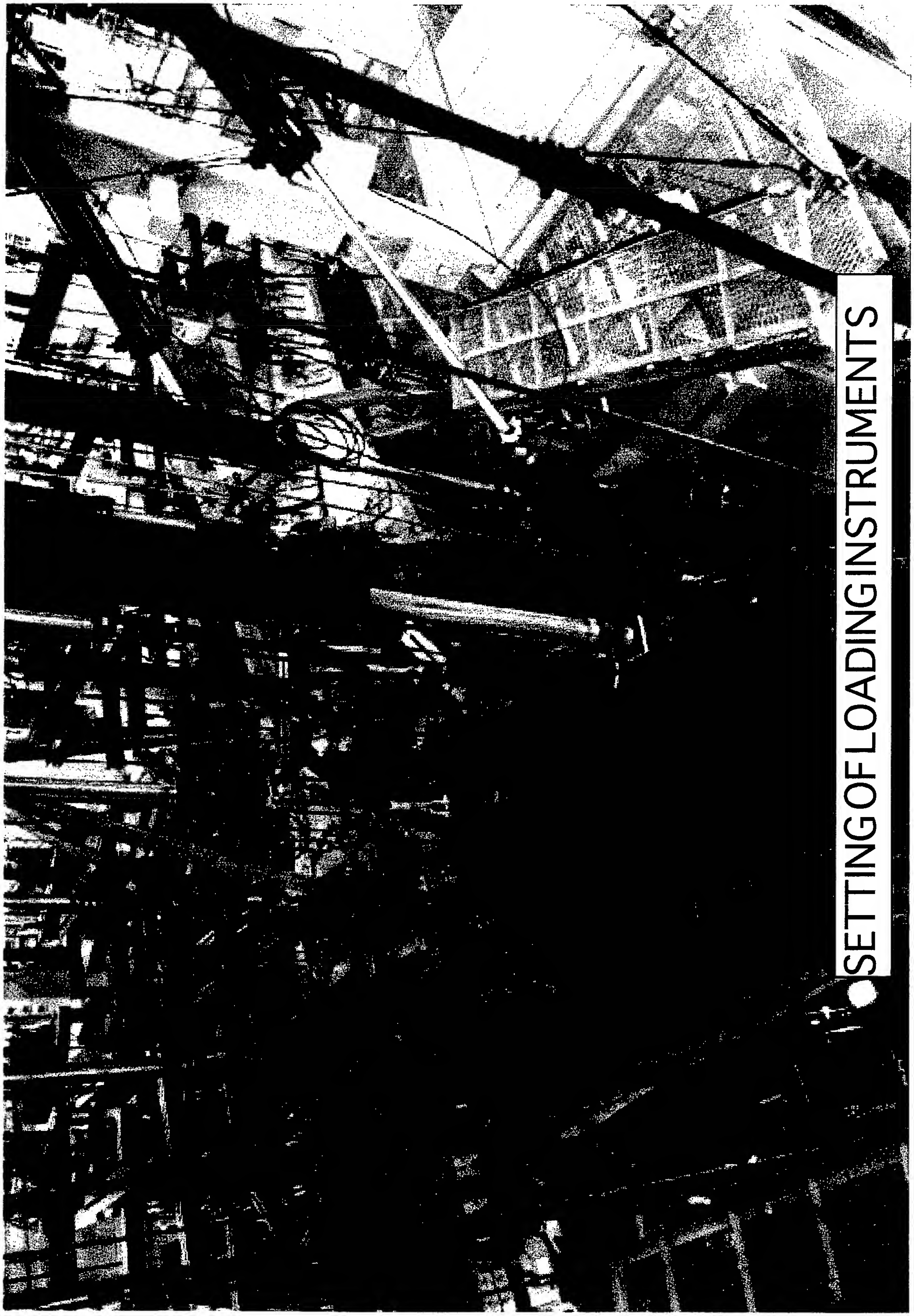
SUMMARY OF TEST ARTICLES




SINGLESEATAIRFRAME


HORIZONTALTAILCOMPONENT


TWOSEATFWDFUSCOMPONENT

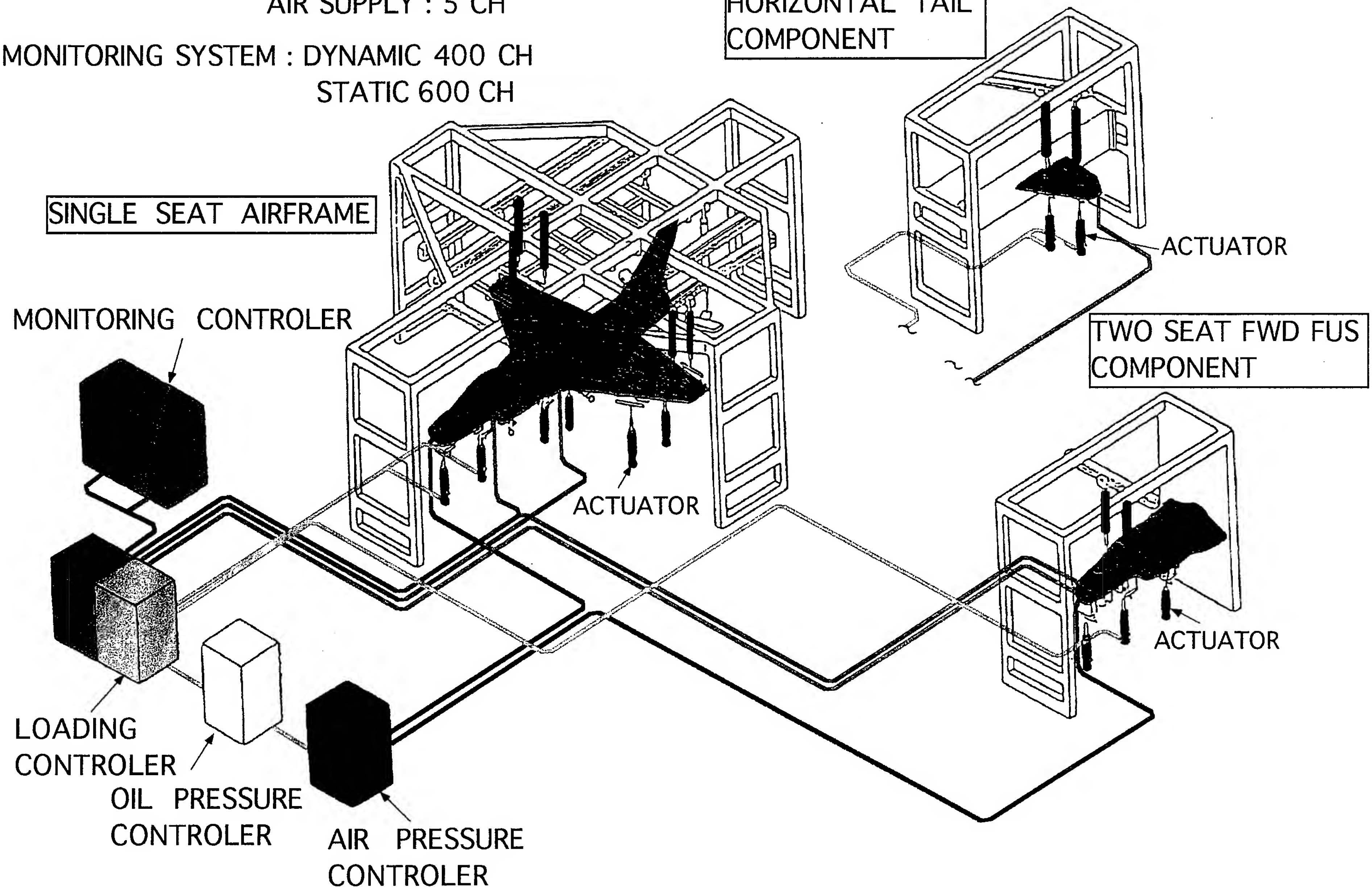


SETTING OF LOADING INSTRUMENTS

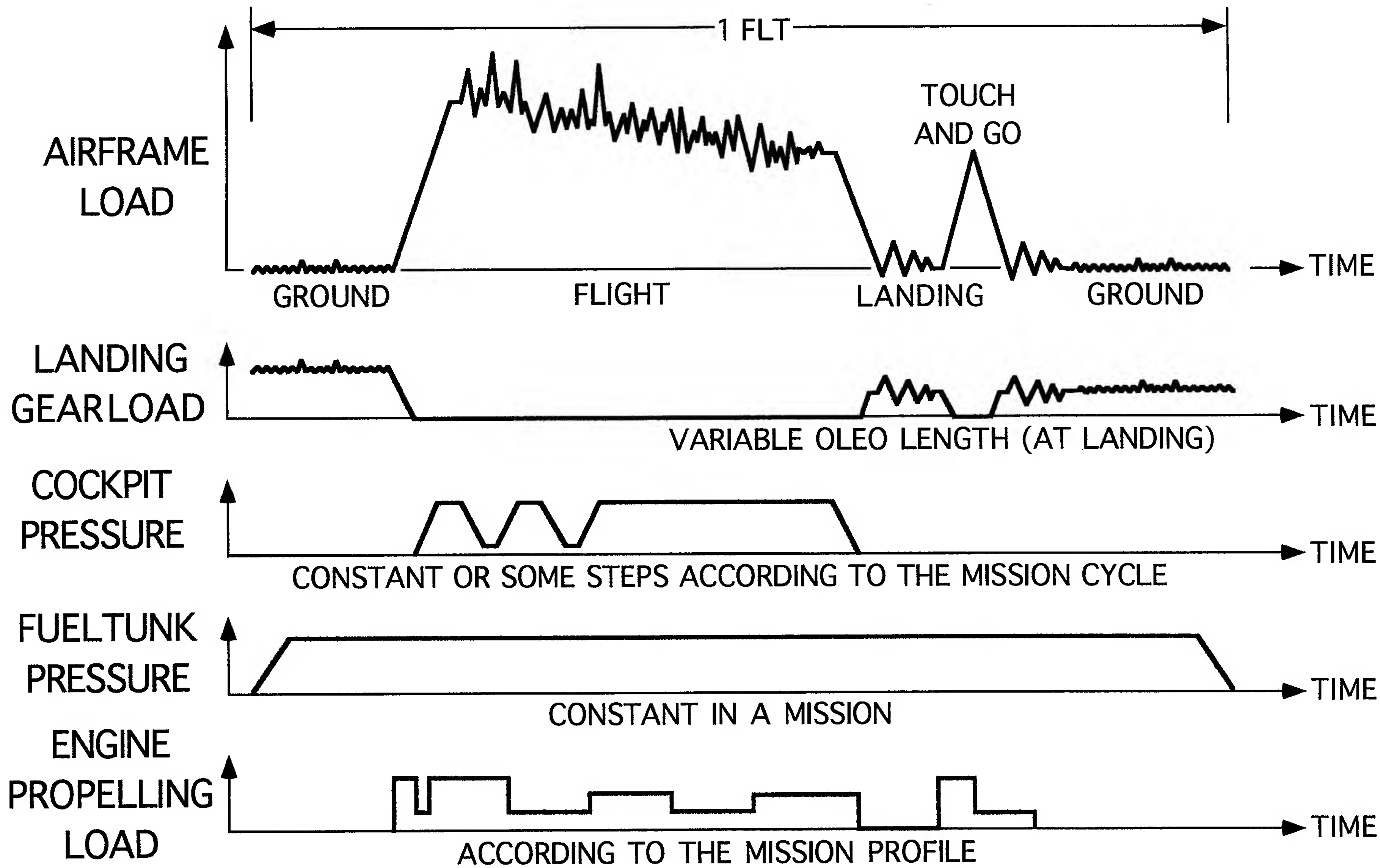
SUMMARYOFTESTINGSYSTEM

LOADING SYSTEM : ACTUATOR : 133 CH
AIR SUPPLY : 5 CH

MONITORING SYSTEM : DYNAMIC 400 CH
STATIC 600 CH

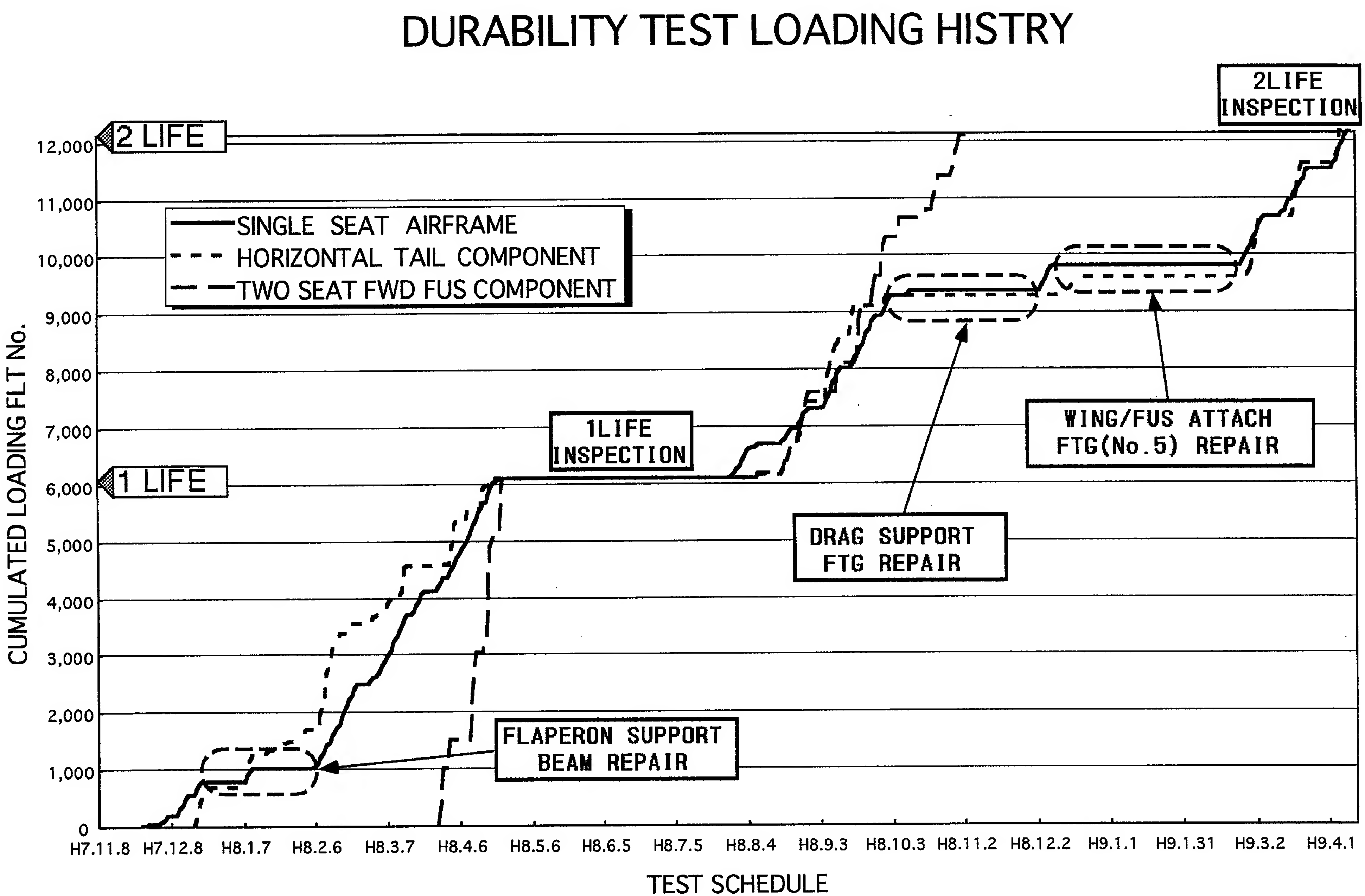


LOADING PATTERN



INSPECTION

CLASS	TERM	CONTENTS
DAILY	1 HOUR	VISUAL CHECK
1／4 LIFE INTERVAL	5 DAYS	VISUAL CHECK(TAKING OFF THE REMOVABLE PANELS)
1／2 LIFE INTERVAL	7 DAYS	AFTER 1／4 LIFE INSPECTION + EDDY CURENT INSPECTION
AFTER 1 LIFE	2 MONTHS	① VISUAL CHECK(TAKING OFF THE REMOVABLE INSTRUMENTS, DUMMY, DOOR, RUDDER, FLAP AND CANOPY) ②ULTRA SONIC, EDDY CURENT INSPECTION AND DYE CHECK ③ALIGNMENT CHECK ④MAIN LANDING GEAR OVERHAUL
AFTER 2 LIFE	3 MONTHS	AFTER 1 LIFE INSPECTION + TAKING OFF THE WING, VERTICAL STAB.,AND DUMMY ACTUATOR
AFTER ALL LOADING (TEARDOWN)	2 MONTHS	①VISUAL CHECK(TEARDOWN) ②N.D.I. (LOCATIONS THAT CRACKS INITIATED OR MAJOR COMPONENT STRUCTURES) ③CUTING OFF THE CRACKED AREA AND SURVEY



DURABILITY TEST LOADING HISTRY
2LIFE INSPECTION
2 LIFE
1 LIFE
1LIFE INSPECTION
SINGLE SEAT AIRFRAME
HORIZONTAL TAIL COMPONENT
TWO SEAT FWD FUS COMPONENT
WING/FUS ATTACH FTG(No.5) REPAIR
DRAG SUPPORT FTG REPAIR
FLAPERON SUPPORT BEAM REPAIR
CUMULATED LOADING FLT No.
12,000
11,000
10,000
9,000
8,000
7,000
6,000
5,000
4,000
3,000
2,000
1,000
0
H7.11.8 H7.12.8 H8.1.7 H8.2.6 H8.3.7 H8.4.6 H8.5.6 H8.6.5 H8.7.5 H8.8.4 H8.9.3 H8.10.3 H8.11.2 H8.12.2 H9.1.1 H9.1.31 H9.3.2 H9.4.1
TEST SCHEDULE

CONCLUSION

- DURABILITY TEST WAS COMPLETED.(2 LIVES OF DESIGN LOAD SPECTRUM)
- CRITICAL POINTS THAT WERE NOT RECOGNIZED BY ANALYSIS OR SUB-STRUCTURE TEST WERE FOUND.
- THE DATA FOR FORCE MAINTENANCE AND MODIFICATION WAS OBTAINED.
- AFTER 2 LIFE INSPECTION, DAMAGE TOLERANCE TEST WILL BE STARTED IMMEDIATELY.

1997 ASIP Conference

Using Emerging Computer Hardware, Software and Communication Technologies in Fleet Management

ROBERT GIESE
GRANT HERRING
OO-ALC/LACM

This paper provides an overview of FLEETLIFE software. FLEETLIFE is aircraft structural management software. The current status, and where emerging technologies should be incorporated to enhance the capabilities are discussed.

Overview

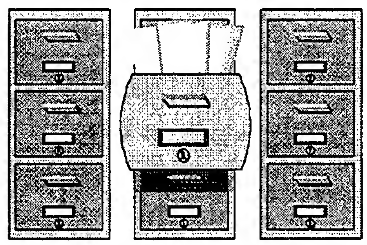
- Data Content
- Data Manipulation and Storage
- Using the Data
- Communications Issues
- Software Requirements
- Putting it all Together
- Damage Evaluation Example

This discussion includes several topics, including the following: what data is necessary for structural fleet management. Once this data content is determined, how do we deal with the large amount of information involved?

One aspect of this problem may be described as not being able to see the forest for the trees. Large amounts of data by themselves are not useful. Effective management and searching and displaying methodologies are necessary to manage an information base and get the most from it. The usefulness of the information in a database is related to the currency of the data. Old or historic data is useful but predictions and fleet planning must be based on current fleet information. Gathering and inputting current fleet information must make use of modern electronic communications technologies. These technologies may include the inter/intranet, satellite communication methods, digital imaging, and standardized software interfaces.. By combining all the emerging electronic information handling technologies and controlling them with a modern effective software data management package, effective and timely fleet decisions can be made.

ASIP Data Content

Fleet Management Data Content



Historic Data

In-Service Data

Fleet management data is necessary to support the Aircraft Structural Integrity Program. These data naturally divide into Historic and In-service categories. Historic data, sometimes referred to a baseline data, is generally derived by the OEM. It forms the basis for the fleet management program. This historic data is continually verified or updated by the collection of in-service data. As time goes on, the quantity of in-service data becomes much larger that the historic data. Fleet trends, structural life extension programs, and structural modification decisions become more a function of relatively recent in-service data. Historic data is always necessary to determine the assumptions and underlying reasons for accomplishing structural actions.

ASIP Historic Data

- Static Strength Analysis
- DTA Analysis
- Full Scale Test Results
- Coupon Test Results
- Original Spectrum Development

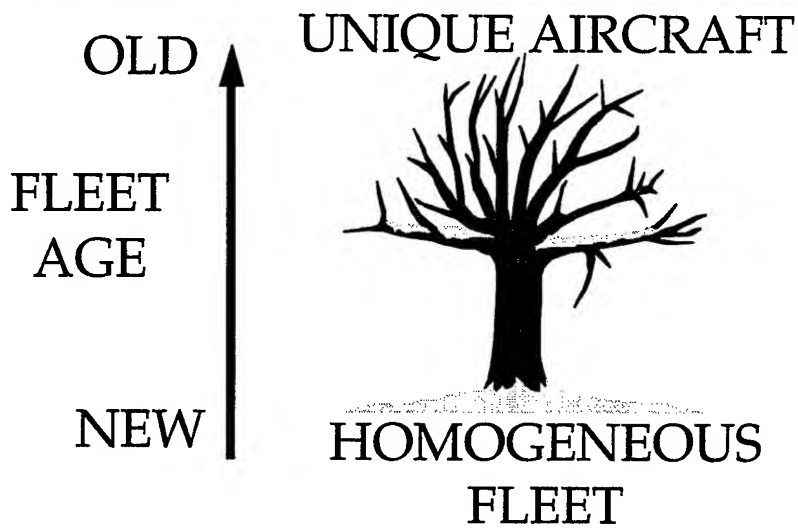
The historic data consists of the original design information. The original design is validated by static strength and damage tolerant analytical (DTA) methods. The results of these analytical efforts are verified by full scale and coupon testing. Archiving this information forms the baseline upon which the structural integrity data is built. To accomplish the original DTA analysis a spectrum must be assumed. How far the actual spectrum deviates from the original spectrum has an impact on the validity of the original DTA analysis.

ASIP In-Service Data

- Structural Repairs
- Structural Modifications
- Loads/Environmental Spectra
- PDM Inspection Results
- ACI Inspection Results

The information developed during usage describes how the aircraft are actually flown and what is done to the aircraft during its life. New predictive models may be required after repairs and modifications are incorporated on the AIrcraft. Also, new mission requirements may strongly alter flight spectra. If the spectrum change is severe enough, rerunning existing analyses or new predictive models may be required. Finally, inspection results validate or modify predictive models

Changing Nature of Fleet Data



As time and usage are accumulated on a fleet of aircraft, each individual aircraft becomes more and more unique. When the fleet is new all the aircraft are relatively the same and a fleet data concept can be used. As the fleet ages many changes are applied to every aircraft. In many cases, unique changes are made to an individual aircraft. Some of these changes are repairs, modifications, mishap damage, combat damage, and environmental effects. This tends to individualize each aircraft and the data content associated with it. Because of this effect, monitoring each aircraft individually becomes more important as the fleet ages. Original fleet data tends to evolve into individual aircraft data.

Data Manipulation and Storage

- Archive Fleet Data
- Link Information
- Allow Sorting & Slicing
- Allow Simple Retrieval
- Provide for Effective and Meaningful Display

The quantity of data generated by a fleet of aircraft is very large. Dealing with this amount of information is a formidable task. Until efficient database management software emerged, the task was impossible. Fleet information, whether stored electronically or on paper, was simply stored. Old data was rarely exhaustively searched and retrieved for gathering information to make a fleet structural integrity decision. At that time the fleet data was not very useful because it depended on the senior engineers to remember where to go to look for information. The data became more and more useless as time and personnel passed.

The first step in efficient fleet management is to archive data in a coherent searchable manner. Certain data is far more useful when it is associated with other data. The second step in this process is to electronically link information in a predetermined manner. The database must now contain a system to sort out quantities of related information in a manner determined by the analysts. This process, to be used effectively, must be simple and easy to use. Once the information desired is retrieved, the database system should provide for clear and informative graphs and reports.

Structural Repairs

- Individual Aircraft Data
  - Tail Number
  - Aircraft Home Station
- Fleet Data
- Responsible Engineer
- Background Data
  - Failure Analysis
  - Metallurgical Analysis
  - Repair Drawings
- Static Analysis
- DTA Analysis

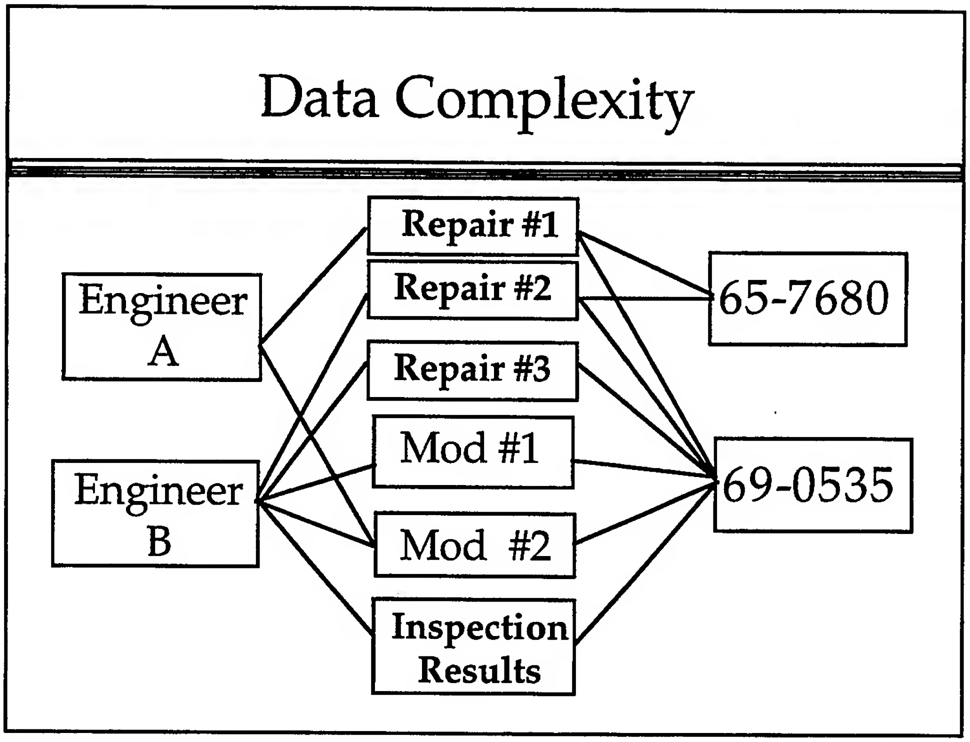
Some examples of the kinds of information are provided in the following pages. The first example is the structural repair. One repair on one aircraft contains a large amount of information. Frequently, a repair is applied to only one aircraft in the fleet. Because of this, the first information recorded is of an archiving nature. The identity of the aircraft, which fleet it belongs to, and who designed the repair must be documented. Next, the information concerning the reason the repair was necessary in the first place is recorded. This information includes any failure or metallurgical analysis information. Repair drawings and analysis information complete the data field. It is also useful in many cases to store digital pictures of the damaged structure. Also, many modern NDE processes generate a digital image or signature. If the repair was necessitated by NDE results, it may be desirable to electronically archive the NDE result with the other repair information.

Structural modifications are also frequently applied to groups or blocks of aircraft in a fleet. The data field in this case consists of slightly different information. First is archiving information concerning which aircraft in the fleet received the modification and when the modification was accomplished on each tail number. Next, the reason the modification was required is documented. Finally, drawings, analyses, and the responsible engineer are recorded.

Results of Inspections

- Aircraft Data
  - Tail number
  - Time of Inspection
  - Base Location
- Fleet Background Data
- Defect Information
- Action Taken

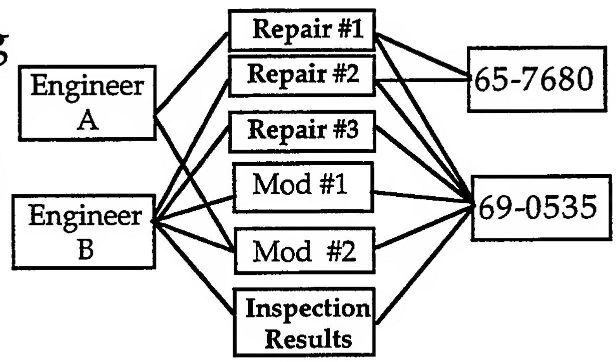
Programmed Depot Maintenance and other inspections are deemed necessary for a variety of reasons. The results of these inspections can be useful for fleet management purposes. Inspection results become very useful when accumulated and stored in an efficient database. Inspection result data can be used to validate or modify analyses. Inspection data can also be useful to detect fleet trends. Trends in fatigue cracking, corrosion, or other unpredicted cracking are some things to look for in inspection data. These inspection data can also be correlated to fleet data. For example, is a specific block of aircraft or a particular structural detail more or less susceptible to trends?



As stated earlier, the amount of data generated by a fleet of aircraft in service is enormous. In this example, there are two engineers working with two aircraft. Repairs may be applied to single aircraft, groups of aircraft, or incorporated into a structural modification. One or both engineers may work independently or together on a repair. Add in modifications and inspection results and someone to monitor those results and the number of links with information becomes very large very quickly. However, it is the integrity of these data links that provides real meaning and usefulness to all of the data.

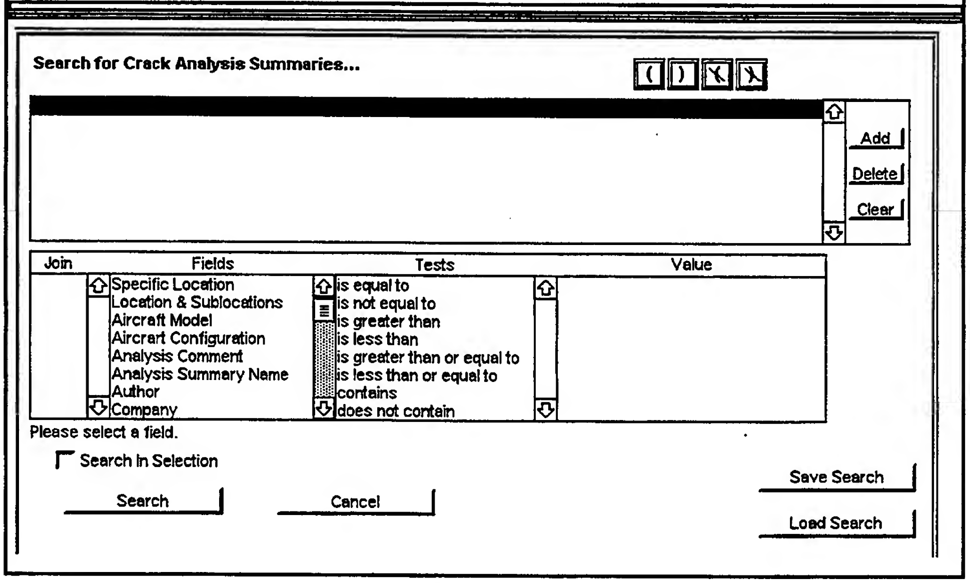
Information Management

- Sorting
- Grouping
- Associating
- Plotting
- Displaying



To make fleet decisions the information or data must be managed and grouped in an attempt to find answers to difficult questions. In this example, Engineer B and Repair #2 are associated together with aircraft 65-7680. Once linked and grouped, the data can be sorted. One possibility in this example is that data could be sorted to determine everything accomplished to aircraft 65-7680. Another possibility is that all of the repairs accomplished in a particular year could be grouped to see the repair activity in that year.

Data Retrieval



In order for data to be useful, specific information must be easily retrievable. To accomplish this, a robust search engine is necessary. In FLEETLIFE software, all key data fields are searchable. Once a data topic or field is determined, several tests are allowed. For example, if a problem developed in the centerline rib in the aircraft it may be desirable to search the database for previous problems and analyses specific to this structural detail or location. So you would search for a specific location equal to centerline rib. Also, perhaps the problem appears specific to one aircraft only. In that case the database could be searched for all information pertaining to that tail number.

Tracking Aircraft

FLEETLIFE Record Finder

67 of 67 records showing | Aircraft Tracking | Recall | No Sets Selected

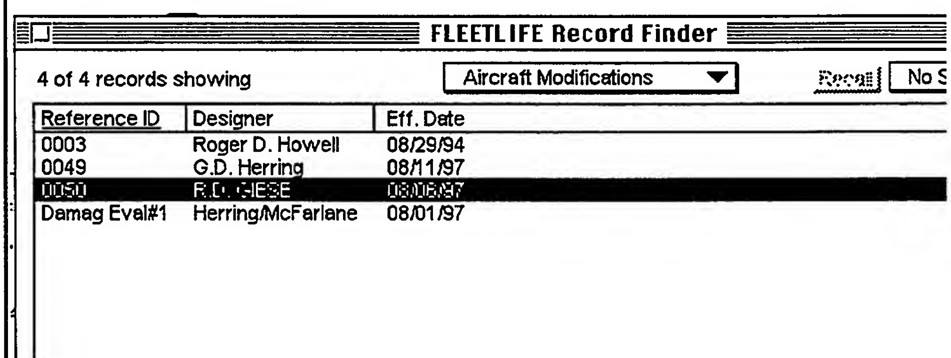
Tail #	Serial Number	Fleet	Model	Base
67-0349	AF119	DRONE UNRESTRICTED	QF-4E Slatted, Thin Skin	Holloman
67-0356	AF127	DRONE UNRESTRICTED	QF-4E Slatted, Thin Skin	TYNDALL
67-0390	AF126		QF-4E Slatted, Thin Skin	Holloman
68-0317	AF129	DRONE UNRESTRICTED	QF-4E Slatted, Thin Skin	TYNDALL
68-0320	AF123		QF-4E Slatted, Thin Skin	Holloman
68-0338	AF115	DRONE UNRESTRICTED	QF-4E Slatted, Thin Skin	TYNDALL
68-0340	AF124	DRONE UNRESTRICTED	QF-4E Slatted, Thin Skin	TYNDALL
68-0342	AF116	DRONE UNRESTRICTED	QF-4E Slatted, Thin Skin	Holloman
68-0343	AF125	DRONE UNRESTRICTED	QF-4E Slatted, Thin Skin	TYNDALL
68-0345	AF101	DRONE UNRESTRICTED	QF-4E Slatted, Thin Skin	TYNDALL
68-0383	AF102		QRF-4C Non-slatted, Thin Skin	TYNDALL
68-0385	AF128		QF-4E Slatted, Thin Skin	Holloman
68-0389	AF130		QF-4E Slatted, Thin Skin	TYNDALL
68-0391	AF122		QF-4E Slatted, Thin Skin	Holloman
68-0449	AF106		QF-4E Slatted, Thin Skin	TYNDALL
68-0555	AF113	Terminated Acft	QRF-4C Non-slatted, Thin Skin	
68-0584	AF110		QRF-4C Non-slatted, Thin Skin	TYNDALL
69-0241	AF133		QF-4G Slatted, Thin Skin	TYNDALL
69-0243	AF143		QF-4G Slatted, Thin Skin	TYNDALL
69-0247	AF136		QF-4G Slatted, Thin Skin	TYNDALL
69-0248	AF140		QF-4G Slatted, Thin Skin	Holloman

Open Record | Search By Location | Search By Value

New Record | Sync Files with Location | Show All

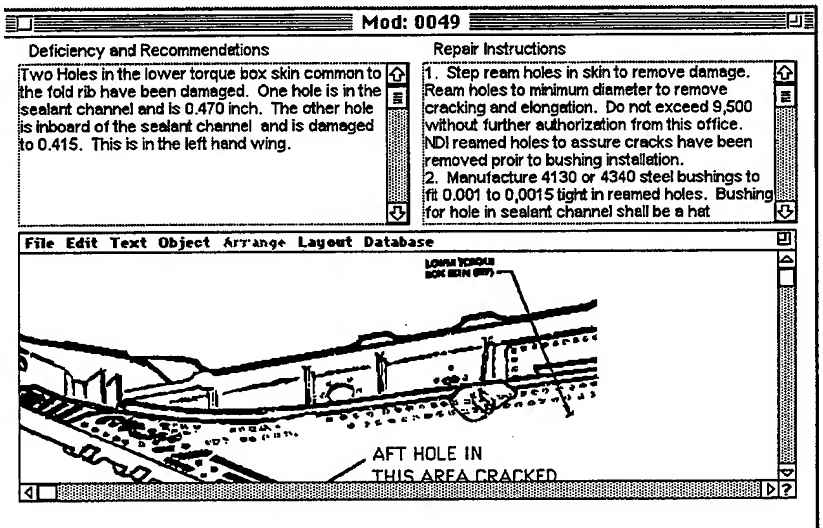
The database stores records. These records have various values or files associated with each record. In this example the data is associated with a tail number. The record as a whole becomes the information associated with that tail number. In this case aircraft 64-0349 with serial number AF119 belongs to the Drone Unrestricted fleet. It is a slatted, thin skin, F-4E based at Holloman AFB. There may also be other information associated with this record. By selecting this record, the information linked to it can be viewed.

Repair Tracking



The record containing the information about individual repairs is keyed to a reference ID number. The top level of a repair record contains the reference ID number, the designer, and the effective date. By selecting a record and opening it, the other information associated with the repair can be viewed.

Repair Documentation



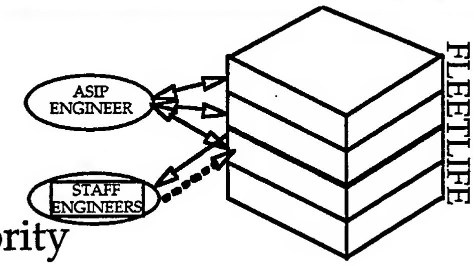
The repair record contains the details associated with a repair. In this example we see the three windows that contain the repair details. The first of these windows describes the deficiencies or damage that necessitated the repair. The next window contains the instructions for accomplishing the repair. The lower window is a simple drawing package. Electronic images of the damage can be pasted into this window. Also, a drawing of the details of the repair can be created directly in this window.

Using the Data

- Who Uses the Data?
- How is it Used?
- Benefits Derived

In the following discussion, who and how the data is used will be discussed. More importantly, what benefits are derived from all this information.

Who Uses the Data?

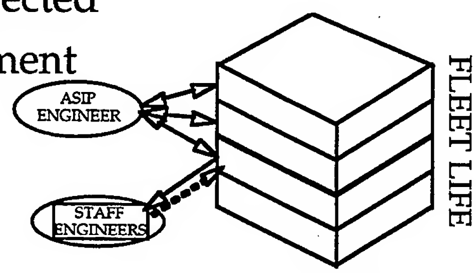


- Fleet Manager
- ASIP Manager
  - Central Key Authority
  - Single Point of Contact
  - Responsible for Database
  - Provides Guidance to Staff Engineers
  - Responsible for Detecting Trends

The fleet manager has an interest in all the high level information concerning trends, fleet averages, and problems. This information is necessary for overall fleet planning. The ASIP Manager is also concerned with trends. Trends such as reoccurring repairs of a specific structural detail, the appearance of corrosion in previously clean areas or changes in fatigue damage accumulation rates are all of importance to the ASIP Manager. The ASIP Manager also has overall responsibility for the integrity of the database. All problems of structural significance pass through the ASIP Manager. The ASIP Manager is constantly vigilant for emerging problems. These problems may manifest themselves first as trends, either as inspection result data or reoccurring repairs.

How is the Data Used?

- Determine Repair Lives
- Determine Inspection Intervals
- Detect Trends
- Plan for Future Actions
- Watch for the Unexpected
- Plan for Fleet Retirement



The software system is used to analyze structural repairs to determine the useful Damage Tolerant life of the structural detail under consideration. If the life of a repair is not long enough, interim inspections will be required. If the repair is on only one aircraft, the information becomes part of the tail number record file. If a repair is incorporated on many aircraft, the single repair record is linked to many tail number records.

When the data is taken as a whole a picture of the condition of the fleet emerges. This picture is used to determine if modifications are required. In some cases extensive structural improvements are required. This forms the bases for a structural life extension program.

Benefits Derived

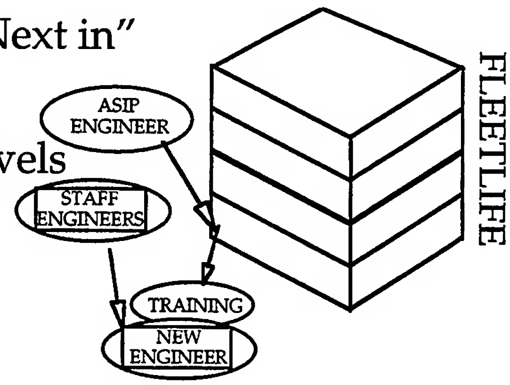
- Continuity of Corporate Knowledge
- Allows Proactive vs. Reactive Fleet Management
- Develops Individual Aircraft Fingerprint
- Provide a Sound Basis for Fleet Decisions
- Allows for Rapid Training of New Engineers/ Analysts
- Creates Clear, Concise Reports

One of the most important benefits to this system is the continuity of the corporate knowledge base. Less reliance is placed on the human memory to remember where to go to look for fleet information. Also, newer staff engineers can use the search engine to look for old records. A robust database can be used to look for fleet trends. Trends can be examined to look for possibilities for extrapolation. Predictive models can be derived from these extrapolations. By accumulating all actions taken on an individual aircraft, its unique structural fingerprint emerges over time.

Finally information not well presented is ineffective. The system allows clear, concise and detailed reports to be created.

DTA Training

- Provides for Corporate Knowledge Continuity
- "Last in" Trains "Next in"
- Sensitivity Studies
- Varying Access Levels



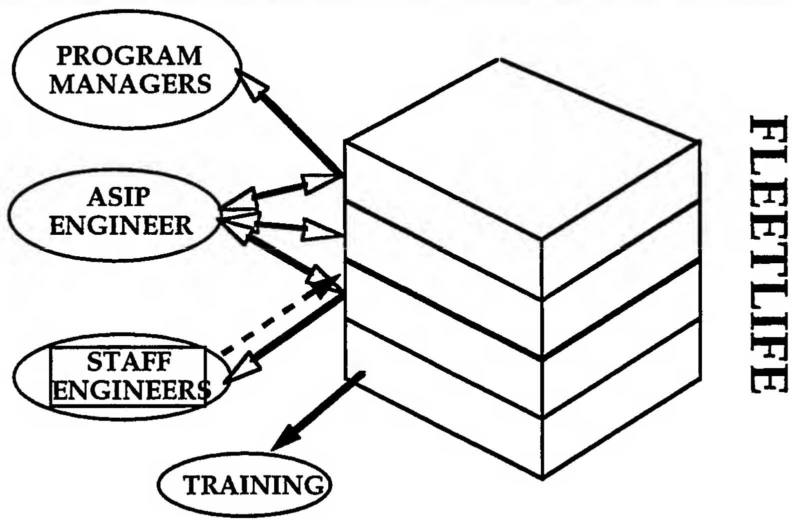
The process of training incoming engineers also contributes the continuity of the corporate knowledge base. An effective method to train new engineers is for the last engineer to enter a group to train the next person to enter. In this manner information and methodologies are continually passed along. Another training method is to accomplish sensitivity studies in analytical methods. This allows new engineers to see the effect of errors in various parameters. This is easily accomplished with a user friendly analysis system.

Communication Issues

- Access to Information
- Security Of the Information
- Incorporating New Information

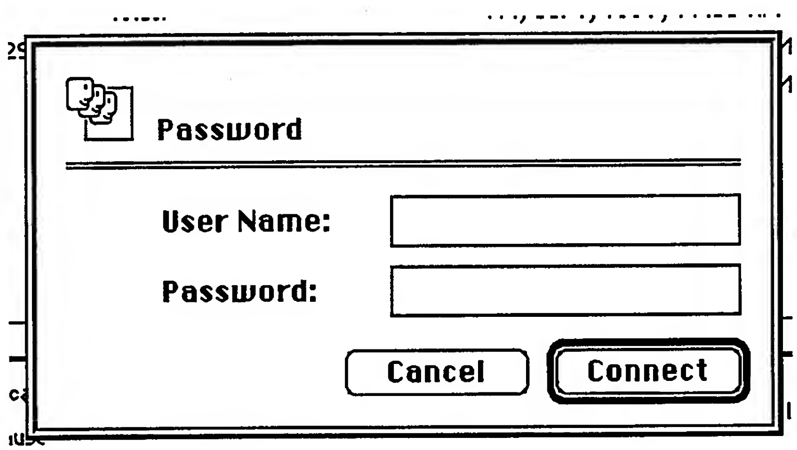
There are several issues of major concern with the interchange of information contained in the database. First is who can access the database. Next is how to prevent contamination of the information in the database. Finally, how to facilitate the addition of useful information in to the system. The following discussion will address these topics.

Access Matches Need



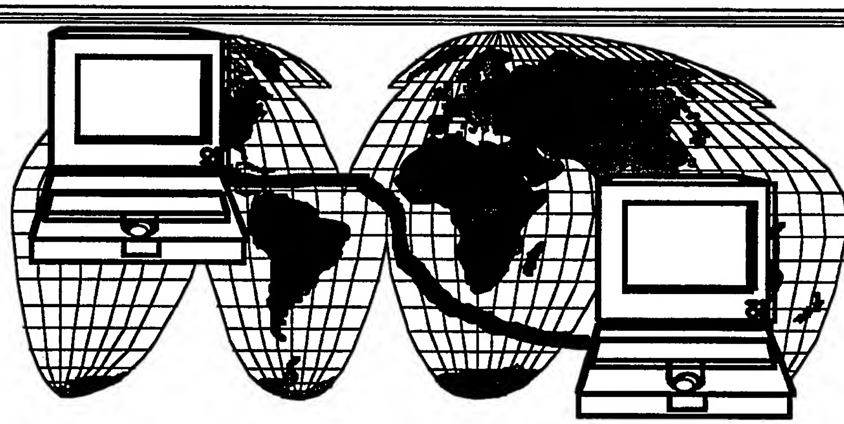
To protect the integrity of the contents of the database, control over access to it is important. Program managers are mostly interested in viewing information from the database. The ASIP engineer is the only person with the access privilege to permanently modify or add to the historic or permanent portion of the database. Other clients can use and add to scratch portions of the database during their day-to-day analytical work. It is then up to the ASIP engineer to decide if this information is added to the permanent database record. Training is conducted in a separate segment of the database. It allows view only of the permanent records. All training scratch records are stored in a separate account.

Password/Level Protection



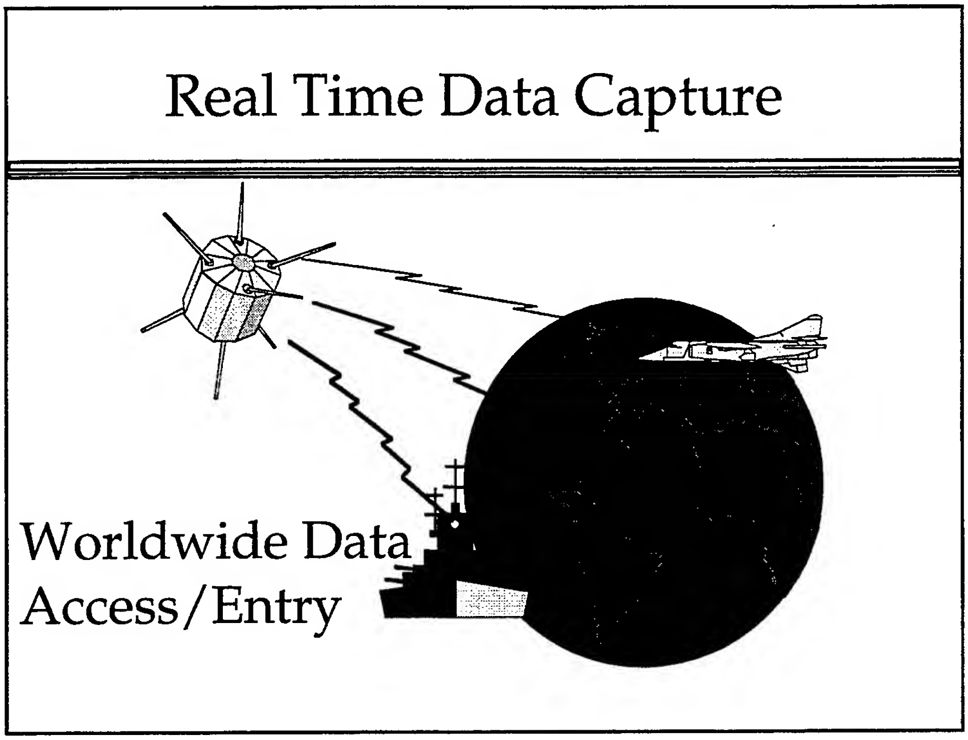
Database protection is provided by the standard password protection method. Special accounts such as an administrator account are also used.

World-Wide Access



Internet Worldwide
User Access

To be most effective, the information in the fleet management database must be as current as possible. Information comes from aircraft stationed worldwide. The database management must allow access and downloading of information worldwide. An effective means to accomplish this is through the internet.



Data integrity is also of importance. In the example, flight loads data is collected worldwide. Presently this data passes through many data processing steps to get to the ASIP engineer. If flight loads data were electronically transmitted frequently or even real time the data capture rates would improve substantially and the error rate would be greatly reduced.

Software Requirements

- Efficient Data Management
- Useful to a Variety of Fleet/ASIP Managers
- Modularized
- Customizable
- Expandable
- Easy to Use

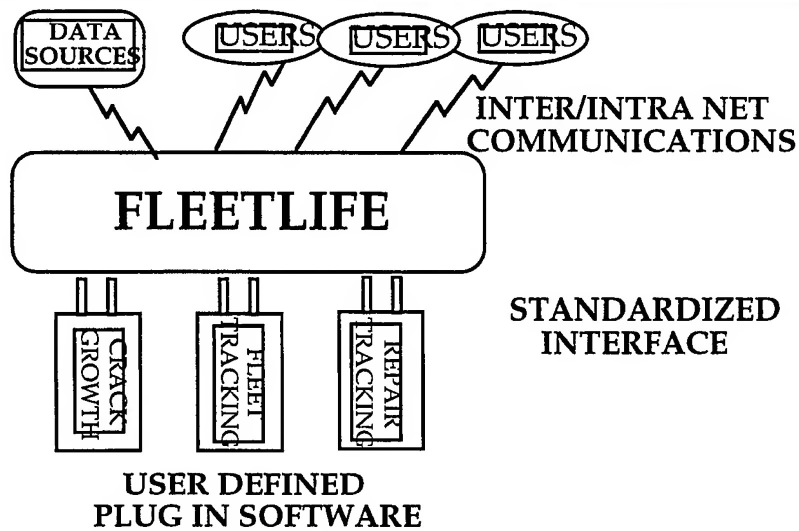
Since the data management software is the heart of this system, several key requirements must be addressed. The software must be as aircraft independent as possible. It should allow for relatively easy expansion of capabilities. It should allow easy tailoring to a specific aircraft fleet. Above all, it should allow engineers to be engineers and not computer experts.

The Goal

Flexible/Modularized Software with Efficient Data Gathering Technology

The overriding goal of this software development effort has been to provide the engineer with a useful ASIP management system. To accomplish this the software system must be flexible to accommodate different models and types of aircraft. It must be modularized so that it does not become so large as to become unwieldy. Only those modules required by a particular user need be loaded on the machine. And finally, the system must easily and efficiently retrieve the information needed.

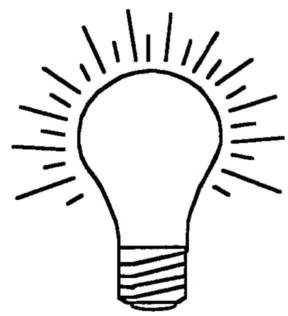
Flexible/Modularized Software



The concept of modularized software with user defined plug-in modules is still in the development phase. To effect this capability there has to be an agreed upon standard interface. There is a great deal of work necessary to make this happen. Use of inter/intra net communications brings along its own set of unresolved problems. The most significant is data security. This is a major issue that must addressed and resolved before effective use of this media can be made.

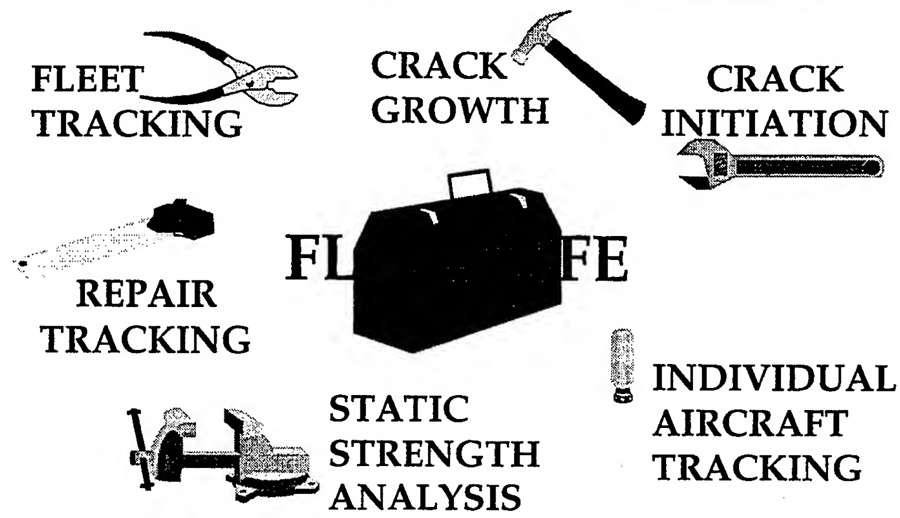
The Fundamental Idea

- Total Package Concept - -
  All essential data and tools for evaluation and documentation of structural life and fleet tracking are shared in one application
- Total User Support Concept- -
  A multi-user, integrated, computation, documentation, and database environment



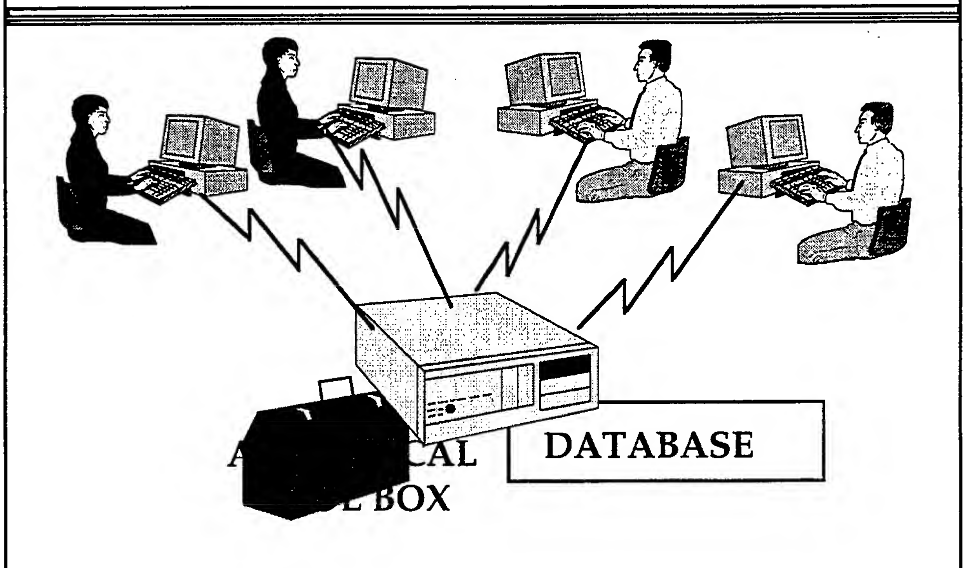
In addition to engineering analysis tools, FLEETLIFE is a powerful data management engine. To provide efficient use by a group of engineers the system provides a multi-user environment.

Total Package Concept



The total package concept involves putting all ASIP support software (tools) into one system. Presently, ASIP engineers rely on several software packages to accomplish their tasks. Some software tools do not at present even exist. Analysis software may reside on several different hardware systems and not allow effective data transfer from one system to another.

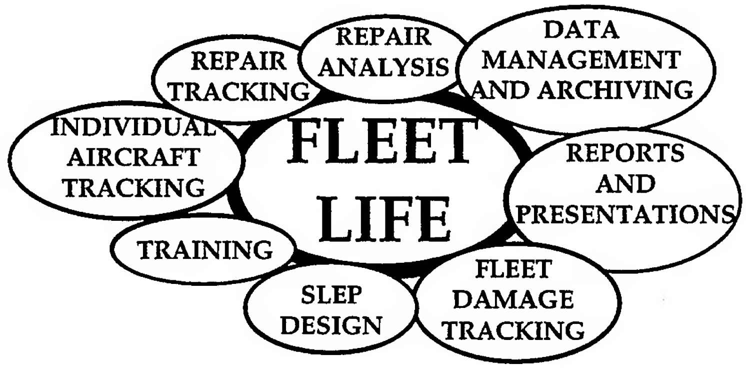
Total User Support Concept



The system must allow multiple users accomplishing several different tasks to operate simultaneously.

The software system must allow multiple processes running simultaneously as well as multiple open windows. The system should support interprocess communications, where one window can automatically update the data in another.

Total ASIP Support Concept



Software Supports The ASIP Functions

The software system is the hub from which all data transfer occurs. In this example the data contained in the fleet damage tracking module and the individual aircraft tracking module are of vital importance the the Structural Life Extension Program (SLEP) design module.

Some of the tools necessary in support of ASIP include crack growth analysis and verification of analysis using test data. To accomplish crack growth analysis a spectrum for the area under consideration must be developed. Also, by studying the flying spectra the fatigue damage accumulation of the fleet can be tracked.

To improve the FLEETLIFE system new ASIP support features are being added. Currently in the prototype phase is the individual aircraft tracking module. Other modules necessary for future development include finite element model results retrieval and static stress analysis tools.

Putting It All Together

- What Needs to Be Developed
- Using Innovative Input Methods
- Software Development Collaboration

Several aspects of this system need to be developed. In addition to software, input devices and methods must be improved. Finally, a method of enhancing the software development is needed.

Future Developments

- Standardized Software Module Interface
- Internet Version
- Worldwide Electronic Data Entry
- Spectrum Manipulation Tools
  - Randomization
  - Rainflow Cycle Counting
- Stress analysis tools
  - Interface with Finite Element Analysis Results
  - Static Strength Analysis Programs

Enhancements to the system are in the planning phases. The enhancements have been identified by the users. To allow user defined software development and incorporation a standardized software interface must be defined and developed. The security of data transmitted over the internet must be addressed. The planned upgrades include additional spectrum manipulation tools. These include randomization and rainflow counting routines.

The static stress analysis tools are based on an accumulation of the structural analysis library of individual analysis tools.

Future Developments

- Additional Crack Models
  - Crack Initiation Model
  - General Crack Growth Model
    - Public Domain Software
- Individual Aircraft Tracking
  - Configuration Management
  - Tracking Modifications and Repairs

To move the FLEETLIFE system into other airframes, other crack growth engines must be incorporated into the system. At the present time the contact stress model is being developed. The contact stress model is used on the F-15 program.

Additional capabilities in the individual aircraft tracking module must be added and evaluated.

Innovative Input Technologies

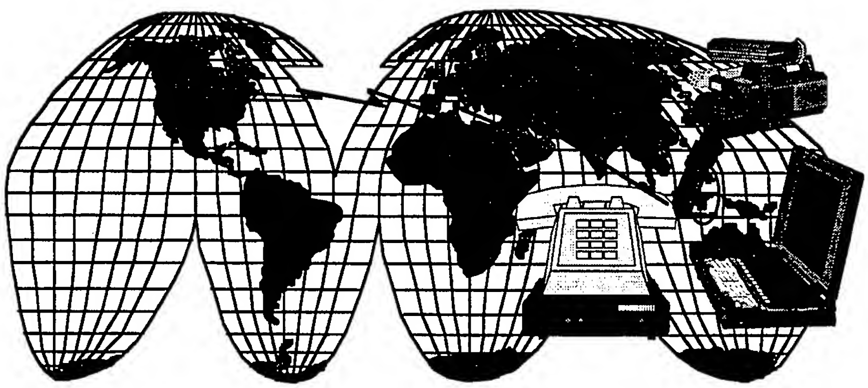
- Digital Cameras
- e-mail
- Inter/Intra Net
- Satellite Communications
- Laptop Damage Evaluation Systems
- Automatic Reporting From Maintenance Database Systems

Innovative input technologies improve the data integrity and capture rates. Both the speed and quality at which fleet information is captured can be improved by using these technologies.

On-Site Damage Evaluation

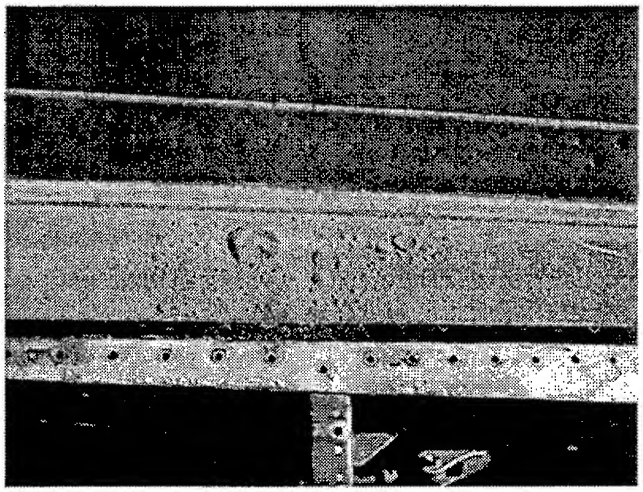
The first improvement in damage evaluation was in the use of electronic media. A typical damage evaluation was accomplished by sending an engineer (or engineers) on site to do the evaluation. Initially this evaluation was accomplished by marking up drawings and notes taken in the field. Photographs were taken and developed on site. Photographs were frequently mailed after the engineer left the station. By using a laptop and a digital camera, all the information required to do the evaluation and design the repairs necessary were easily transportable. It was no longer necessary to develop photographs and ship them.

Remote Damage Evaluation



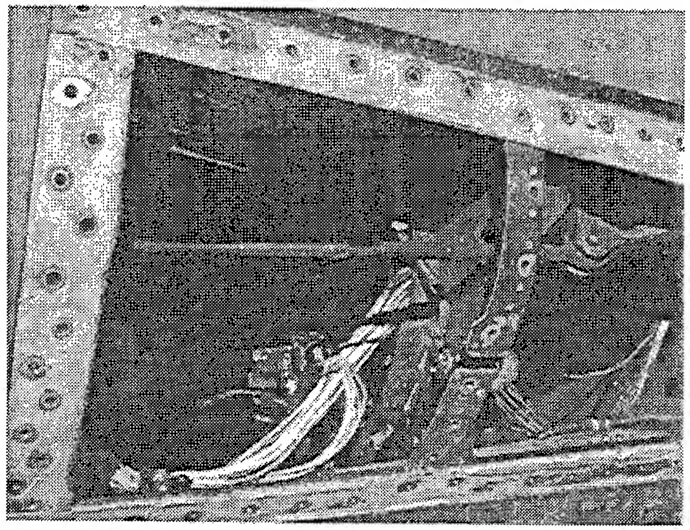
In this case the owning activity uses a digital camera. The engineers doing the evaluation instruct the owning activity on the types of shots needed to do the evaluation. The pictures are then transferred electronically. This is accomplished either by e-mail or with a modem. After the pictures are received and viewed, not all information may be available to do a damage evaluation. If this is the case the using activity is contacted and a new set of pictures made and sent.

Damage Results #1



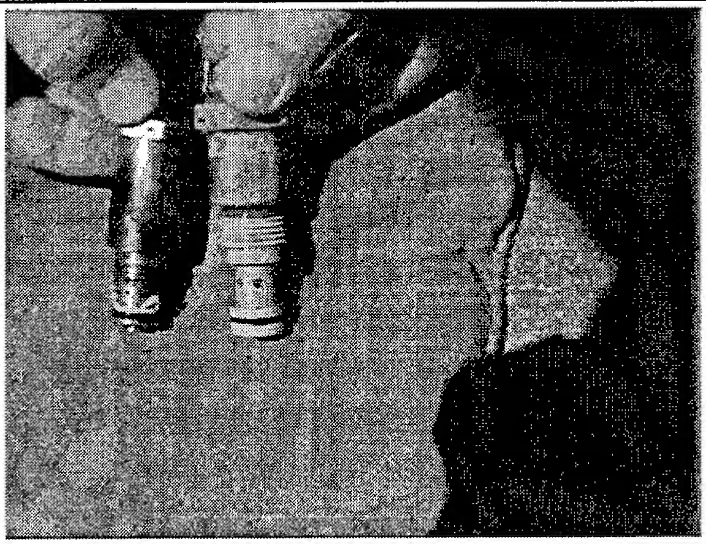
This is an example of burn damage on a skin panel.

Damage Results #2



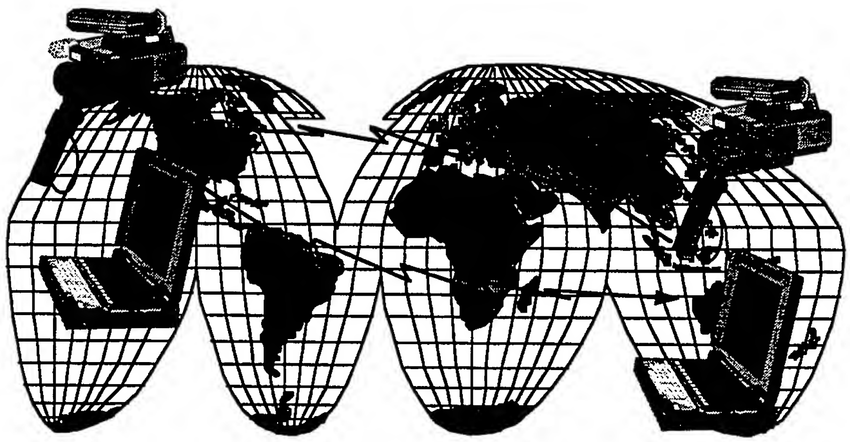
This example shows burned wiring and connectors.

Damage Results #3



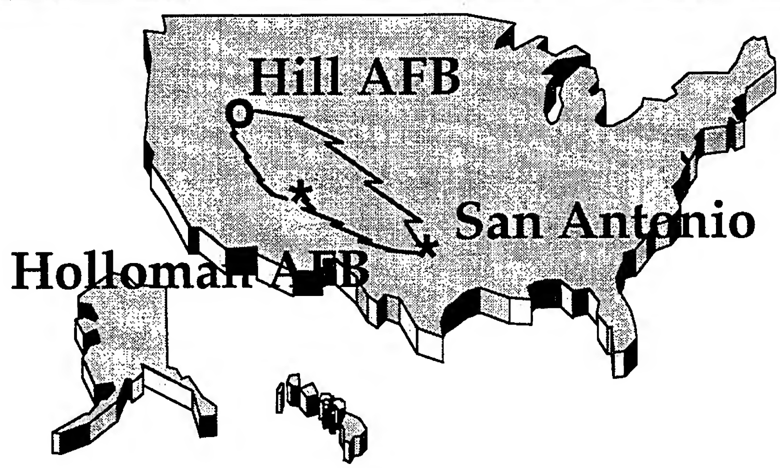
This picture shows two small hydraulic components. The one on the left is heat damaged.

Real Time Interactive Damage Evaluation



Real time damage evaluation has a camera/software/computer at both ends. The computers are connected through telephone lines or perhaps an inter/intranet connection. Voice and video data is continually transmitted. Snapshots for fleet records can be taken from the video stream. Instructions for lighting, camera angles, and distance can be made real time.

Demonstration Concept



In this demonstration a camera/computer set-up is stationed at Holloman AFB. A projector is at San Antonio, TX. Images from Holloman are projected on a screen in San Antonio. Data is also transmitted to the Home Station at Hill AFB, UT. Data is captured and stored in the database at Hill AFB. This demonstrates a scenario of a damage evaluation being conducted while away from home station.

Conclusions

- ASIP Enhanced by an Extensive, Centralized, Fleet Data Repository
- Fleet Management Enhanced by Efficient, Easy to Use Data Management Software
- Capabilities Can Be Greatly Improved by Effective Use of Worldwide Communication Technologies

FLEETLIFE Steering Group

- Chaired by OO-ALC/LACM
  - Jess Young, DSN 777-5291, EMAIL YOUNGJ@HILLWPOS.HILL.AF.MIL
- Current Members
  - FAA Technical Center
  - F-15 ASIP Manager
  - A-10/F-111 ASIP Manager
  - AV-8B In-Service Support Team
  - Southwest Research Institute
  - McDonnell Douglas Aerospace

For additional information about FLEETLIFE or to get involved in the steering group, contact Mr.. Jess Young at the above address.

SESSION II

LIFE ENHANCEMENT

Chairman - ***K. Leikach***
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TECHNOLOGY
INC.

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Terminating Repair or Resizing of Damaged/Discrepant Holes Using Expanded Bushings

By Len Reid, Vice President of Engineering
and
Jude Restis, Engineering Manager
Fatigue Technology Inc.
Seattle WA USA

Paper presented at:
1997 USAF Aircraft Structural Integrity Program Conference
San Antonio TX USA
December 2-4, 1997

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ABSTRACT

Damaged or discrepant holes are common in aircraft production and structural repair or modification programs. Damage induced by drilling or mis-alignment requires oversize of fasteners or in extreme cases bushing or plugging of the hole to reposition it or remove all incipient damage. In repair of aging aircraft, removal of fatigue or corrosion damage in holes necessitates similar oversizing and frequently needs application of splice repairs or component replacement.

This paper discusses the use of cold expanded bushings in repairs. These methods were adopted from split sleeve cold expansion technology and result in a convenient high integrity repair with generally a better fatigue and damage tolerance life than the original structure. Several military and commercial aircraft applications have successfully used these methods as terminating repair solutions. Examples of these applications and supporting test and other data are presented.

INTRODUCTION

Damaged and discrepant holes are a common problem during the production, assembly, modification and repair of aircraft structure. In service, hole damage arises from fatigue, corrosion, impact damage, wear and fretting which is a serious concern to the aging aircraft engineer as it reduces both structural integrity and safety. Discrepancies are generally caused by mis-alignment of drilling devices, riveting guns and fixtures resulting in double drilled holes, off-centered holes, angled holes, etc. Production aircraft discrepant holes may number in the low thousands.

Most damaged or discrepant holes can be repaired by enlarging the hole and installing oversize fasteners; however, these are not always readily available and are generally more costly than nominal fasteners. Using larger fasteners may result in low edge margin or incomplete removal of the discrepancy or damage. They can also locally stiffen the joint causing unanticipated fatigue problems in adjacent areas.

Plugs and shrink fit bushings are used in repairs where oversize fasteners are not suitable, or do not eliminate all incipient damage. The plugs or bushings are good in that they refill the hole and allow the use of nominal fasteners. Although bushing repairs have been used in many demanding applications they usually do not provide a terminating repair and often require repeat inspections and/or follow-on repairs.

Fatigue Technology Inc. (FTI) recently completed a test program comparing the crack growth life of holes repaired with FTI's BushLoc hole repair and resizing method and the ForceTec rivetless nut plate system in a short edge distance application. These cold expanded bushings provide a number of important performance advantages over

shrink or freeze fit bushings that allow them to be used as a terminating repair in many applications. The advantages include: significantly longer fatigue and crack growth lives, higher retention forces, and ease of installation. The results of this recent testing will be discussed along with a number of successful terminating repair actions and other novel repairs using cold expanded bushings.

BACKGROUND

While being a convenient repair method, shrink fit or press fit bushings may not be the best solution when considering the long term implications since many of these repairs are not terminating. Terminating repairs are defined as those repairs with a projected fatigue life significantly longer than the projected service life of the aircraft. In an era of budgetary constraints and limited numbers of new aircraft coming on-line it is essential that repairs on the current fleet be designed with this in mind. Cold expanded bushings provide an effective method of repair that in many instances provide a terminating action as shown in Figure 1. In this hypothetical example the structure requires supplemental inspections or at worst, two repairs on a given location in order to meet the service life goal. Alternatively, the cold expanded bushing repair method requires only a single repair to meet the life objective and possibly elimination of supplemental inspections.

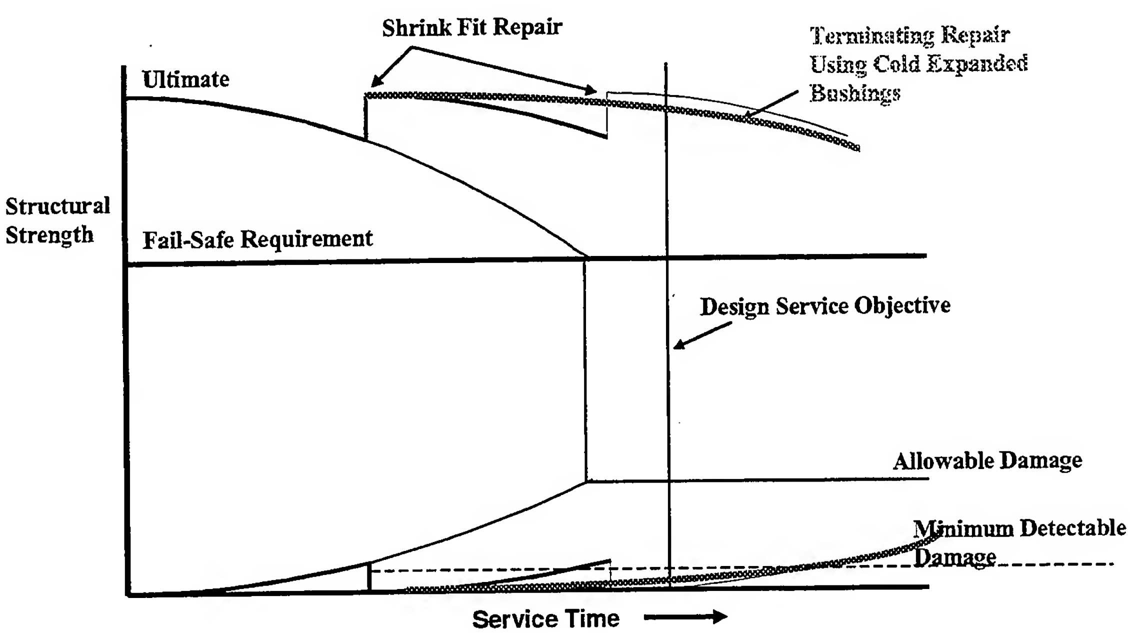


Figure 1: Effect of Terminating Repair on the Damage Tolerance and Service Life of the Aircraft

COLD EXPANDED BUSHING METHODS

There are three methods of installing cold expanded bushings that have been developed by FTI. These methods are:

1. ForceMate® which installs initially clearance fit bushings with high interference fit resulting in significant fatigue life improvement,
2. BushLoc® A convenient repair/resizing bushing with high interference fit using a variation of the split sleeve cold expansion process, and
3. ForceTec® A rivetless nut plate which can be used for repairing fatigue damage associated with conventional riveted nut plate installations.

These will be discussed and examples shown of actual or tested applications on military or commercial aircraft. The merits and benefits of each method will become evident.

SUPERIORITY OF COLD EXPANDED BUSHINGS

Cold expanded bushings are superior to shrink and press fit bushings in many ways. The primary advantage is the fatigue life improvement resulting from the unique state of residual stress around the hole. Increasing the fatigue life reduces the need for frequent inspections and increases the overall integrity of the repair. The typical life improvement, ranging from 3:1 to greater than 20:1, allows the cold expanded bushing to be used as an integral part of a terminating repair.

Fatigue Life Improvement The action of cold expanding the bushing generally imparts compressive residual stresses around the hole, depending on the bushing/parent material combination, that reduce the mean stress at the hole thereby improving fatigue life. Furthermore, the high interference fit of the bushing acts to reduce the stress amplitude at the hole. These two effects work synergistically to significantly improve fatigue and crack growth lives. This is illustrated in Figure 2. These beneficial residual stresses have a profound effect on the resulting fatigue and crack growth life of both new and repair bushing installations as shown in Figure 3.

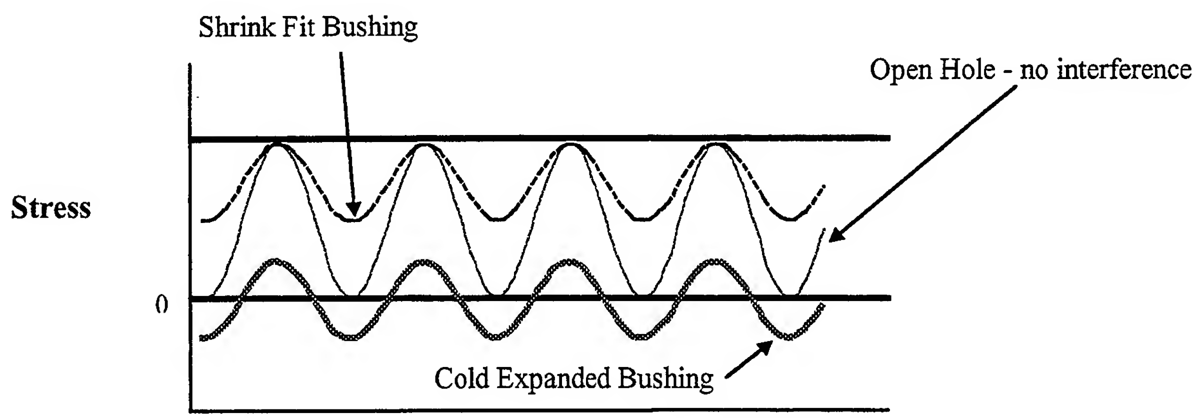


Figure 2: Comparison of Shrink Fit and Cold Expanded Bushings - Cyclic Stress of a Bushed Hole

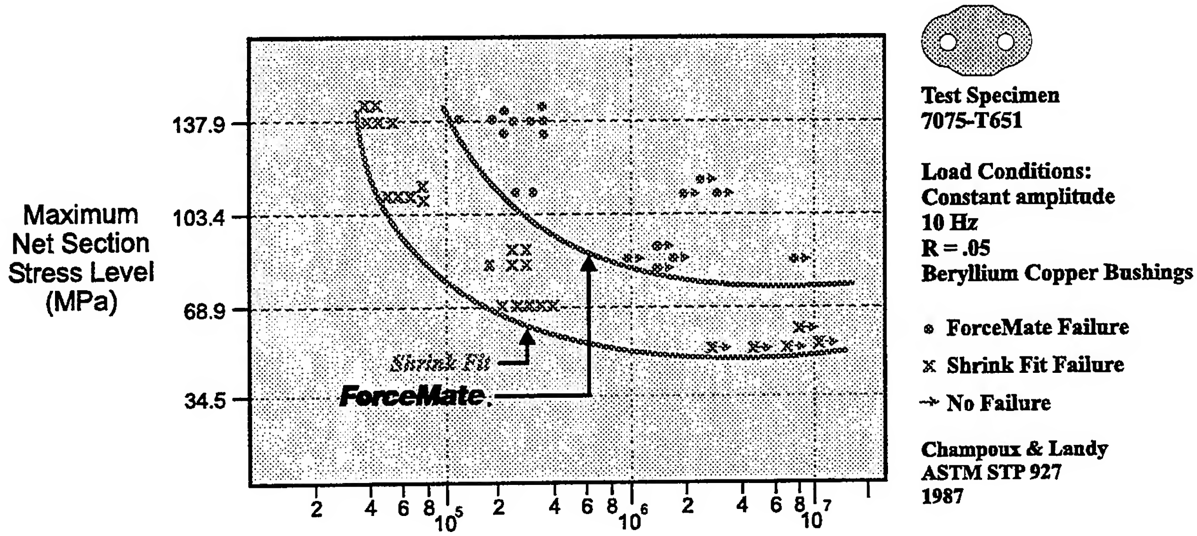


Figure 3: Fatigue Life Comparison of Shrink Fit and ForceMate Bushing Installations

Interference Fit The resulting high interference of each of the bushing installation methods improves resistance to push out and torque significantly. The high resistance to these forces makes the ForceTec rivetless nut plate possible. High retention of cold expanded bushings allow them to be used in areas prone to bushing migration due to vibration or high stresses. Push-out forces are typically doubled when compared to those of a shrink fit bushing installation as shown in Figure 4. The figure shows not only the near doubling of push-out resistance, but shows the consistency of the range of push-out. For the shrink fit bushing push-out ranged from 800 to 2,000 lb. because of the variability of the interference. By comparison the ForceMate installation push-out ranged from 2,600 to 2,900 lb for a 1.0 inch diameter bushing installed in 7075-T73 aluminum.

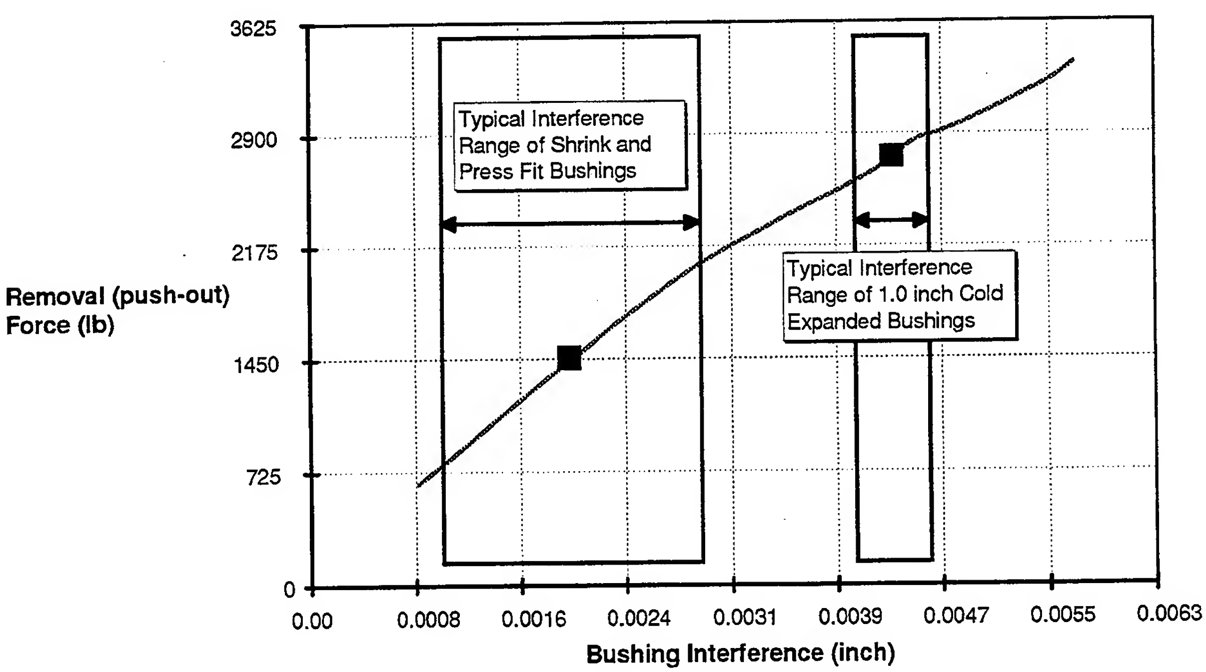


Figure 4: Comparison of Push Out Forces - Shrink Fit Versus ForceMate™

Corrosion Resistance Cold expanded bushings provide better corrosion resistance due to the nature of the installation. Each bushing or retainer is initially installed into the hole with a slight clearance fit. The small gap allows the use of liquid or solid sealants, platings and coatings that can be applied to the bushing without damaging or scraping them off. Shrink or press fit bushings are installed either net or slight interference fit with little or no gap for corrosion preventative compounds or sealants. Protective platings are typically scraped as they are installed into the hole. Additionally, condensation from the shrink fit process remains which can lead to corrosion.

Flexibility of Repair Configurations Each of the cold expansion bushing process comes in a wide variety of diameters and lengths to meet just about any application. ForceMate bushings can be made to meet the most exacting specifications including dimension, tolerance, and material. The BushLoc process offers the greatest flexibility for cold expanded repair bushing installation. Just about any combination of bushing length, inside diameter and outside diameter can be installed with this process. The ForceTec rivetless nut plate process offers retainers in standard and oversize diameters ranging from 3/16 to 1/2 inch with lengths ranging from 0.060 to 1.0 inches in increments of 0.010 inches.

Several different expanded bushing repair configurations have been evaluated. The following examples demonstrate the versatility of the process to either install multiple bushings or repair multi-layered stack-up joints containing fatigue cracks or damage.

Figure 5 shows a multi-layered stack-up with individual segmented bushings. All three bushings are installed simultaneously. In this type of installation, the combinations of bushings and parent materials may be different. Final line reaming of the installed bushings would be required if different combinations were used because the final inside diameters would vary due to the different amounts of "springback" after bushing expansion.

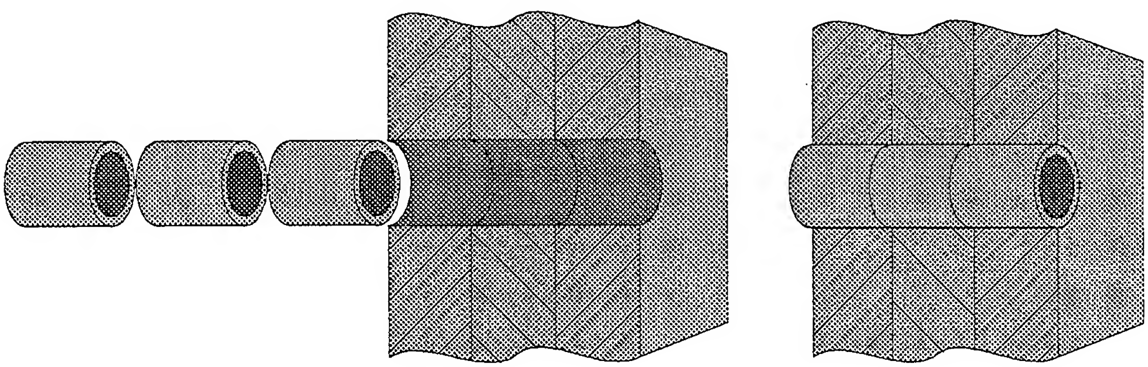


Figure 5: Schematic of Multiple Bushing Installation

In Figure 6, different outside diameter bushings can be simultaneously installed allowing minimum material removal to correct hole discrepancies or to remove corrosion damage or fatigue cracks. This "Christmas tree" arrangement, and the previous multiple bushing installation, can be done without necessarily breaking down the structure to install the bushings.

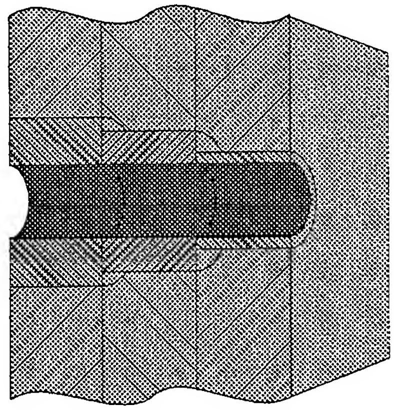


Figure 6: Multiple Variable Wall Thickness Installed Bushings

Repair of Countersunk Holes BushLoc and ForceMate can also be used to repair previously oversized countersunk holes. In this case, a matching countersunk bushing is made to fit the residual countersink in the hole. Figure 7 shows a typical repair situation where two repair bushings are installed in a skin to a sub-structure fastener hole. Either a protruding head or nominal countersunk fastener can be installed. A washer of larger diameter than the outside diameter of the bushing is required under the retaining nut or collar of the installed fastener to secure the bushing and fastener to the structure.

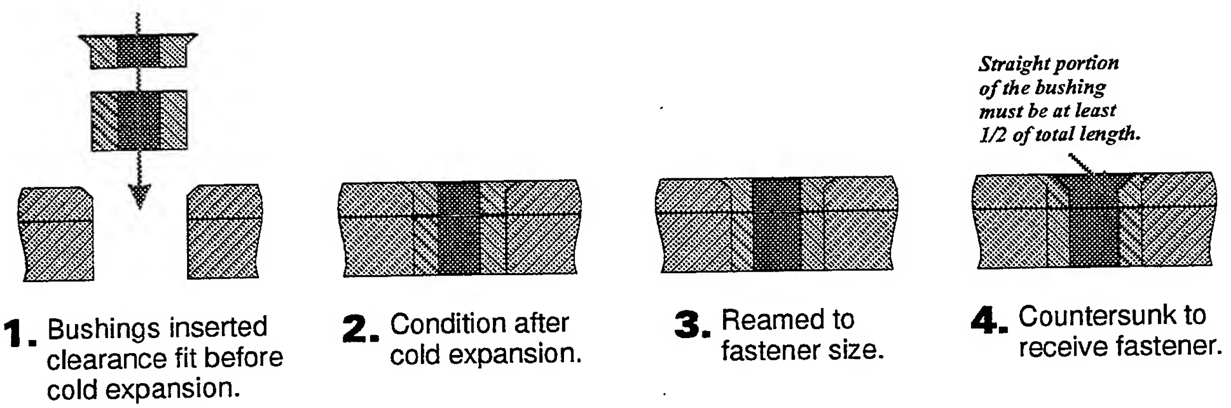


Figure 7: Repair Scenario for Repaired Countersunk Hole

Other advantages of the cold expanded bushing process include;

- Generally wider manufacturing/rework tolerances of holes and repair bushing
- Quick installation
- Commonality of installation tooling

DETAILED PROCESS DESCRIPTIONS AND APPLICATIONS

ForceMate High Interference Fit Bushing Installation Process

ForceMate™, radially expands an initially clearance fit, internally lubricated bushing into the hole using a tapered expansion mandrel as shown in Figure 8. The process simultaneously installs the bushing with a high interference fit [up to 0.010 inches (0.25 mm)] and imparts beneficial residual stresses to the material surrounding the bushing.

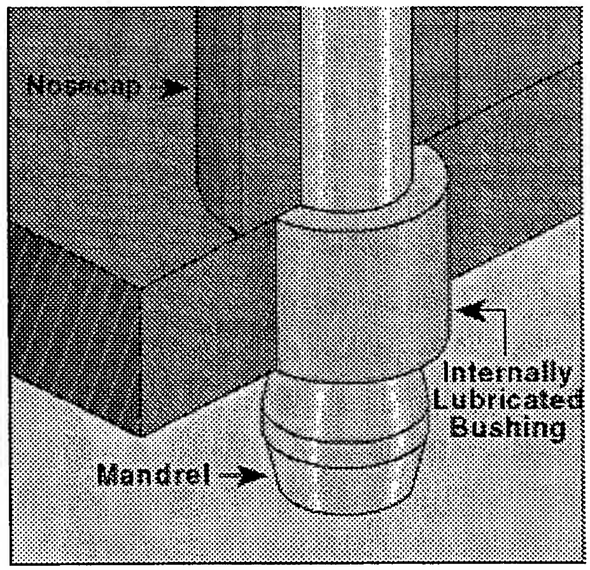


Figure 8: Typical ForceMate™ Process Steps

A comparison of shrink fit and ForceMate residual stresses for a 1.0 inch 4340 steel bushing installed into 7075-T73 aluminum lug with a w/D of 2.5 is shown in Figure 9. Note the residual tensile stresses of the shrink fit method in this example.

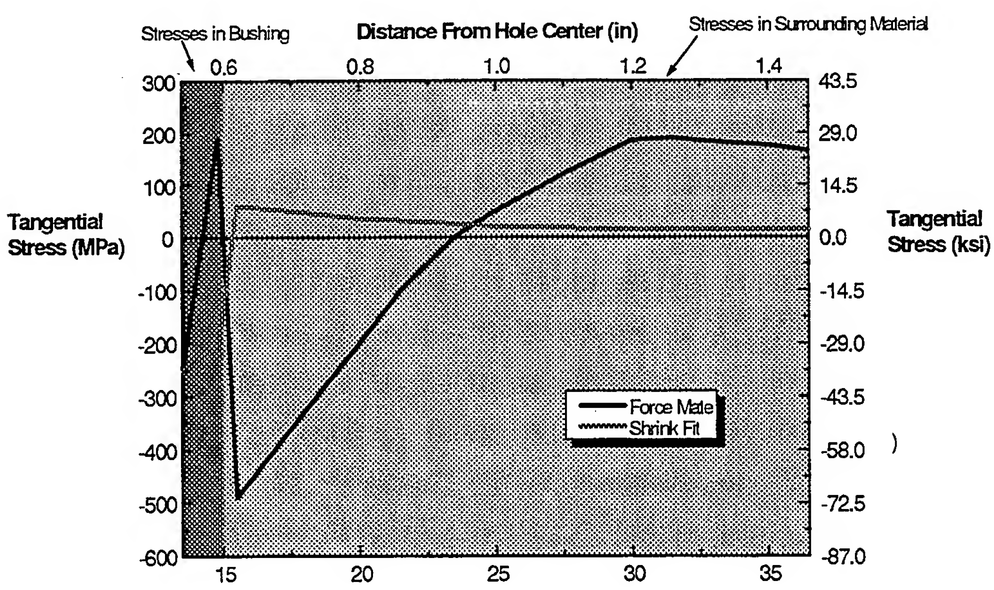


Figure 9: Comparison of Residual Stresses - Shrink Fit Versus ForceMate™

ForceMate Applications

One of the more interesting uses of ForceMate in a terminating repair was for the F-16 Fighter Doors Rework. The concept of this repair was to convert a non-load carrying access panel into a load carrying structural member. By carrying load through the panel the high stresses at fastener holes in the corners of the underlying frame could be reduced thereby increasing fatigue life. An 8,000 hour life goal was the objective of the program.

To make the repair successful three primary components were necessary; metal matrix door, expandable Avibank fasteners and ForceMate bushings. The bushings were used to resize and cold work the nut plate fastener holes in the frames. The expandable fasteners allowed for tight fit of the panel for good load transfer while the new panel offered increased strength and lighter weight than the aluminum panel.

Crack growth testing of the ForceMate repair bushings on subcomponent specimens demonstrated significant life improvement; at least 2:1. Follow on testing on a subcomponent, that better represented the actual doors showed improved life as well; about 2,500 hours for original configuration compared to double that for ForceMate. A similar test of the expandable fasteners on this specimen configuration demonstrated a similar life. While significantly improved, the repair bushings and the fasteners tested individually did not meet the life goal. A further cyclic test was conducted with the combination of the ForceMate bushings and the expandable Avibank fasteners which demonstrated equivalent flight hours in excess of the required hours.

BushLoc™ Bushing Installation Process

The next cold expanded bushing installation method is the BushLoc™ process. The installation of a bushing using BushLoc is accomplished using a specially designed tooling similar to that used to spilt sleeve cold expanded a hole. A pre-lubricated, stainless steel split sleeve is placed over a tapered expansion mandrel, which is in turn attached to a hydraulically operated puller unit. A bushing sized to specific BushLoc dimension, is placed onto the mandrel and over the sleeve, and then placed in the hole, as shown in Figure 10. When the mandrel is drawn through the sleeve, bushing and surrounding metal are subjected to radial expansion forces. The radial expansion and subsequent unloading impart beneficial stresses around the hole much like those created during the ForceMate process. The key differences between BushLoc and ForceMate is that the former uses a split sleeve during the installation process, has relaxed bushing tolerances, and the bushing can be locally manufactured.

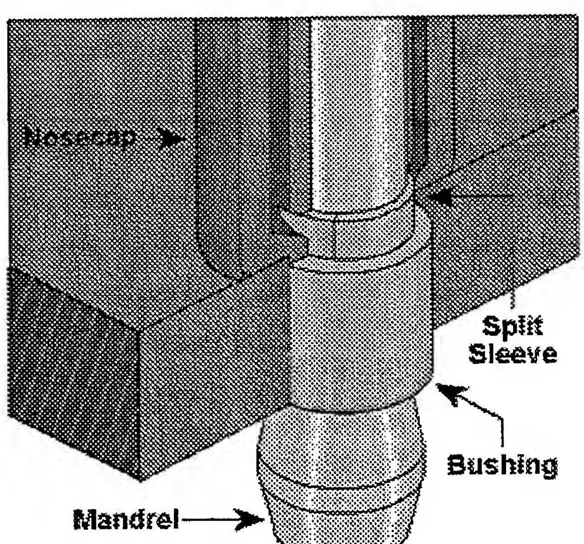


Figure 10: Typical BushLoc Bushing Installation

BushLoc Applications

C-141B Spanwise Splice Repair Evaluation FTI recently completed a test program comparing the spectrum fatigue life of both FTI's BushLoc hole repair and resizing process and the ForceTec Rivetless nut plate system in a short edge distance application. Testing was done at the request of Warner Robins Air Logistics Center's Technology and Industrial Support Directorate as an aging aircraft initiative in support of the C-141B maintenance program.

The spanwise splices of the lower inner wing have been recognized as life limiting locations for the aircraft. The potential for widespread fatigue damage at these locations, as well as limitations in detection of very small cracks which might emanate from spanwise splice fastener holes, raised concern about composite doubler repairs in the vicinity of the fasteners. Increased stresses in the wing planks near the ends of the composite doublers could aggravate the problem. BushLoc and ForceTec were evaluated as possible repairs or preventatives of small cracks at fastener holes near the ends of composite doubler repairs.

The test was conducted on 7075-T6511 pre-cracked, short edge margin aluminum specimens taken from actual C-141 wing structure. The original 1/4 inch diameter [edge margin (e/D) =2.0] holes were pre-notched and the crack grown to a part-through surface length up to 0.070 inches. To accomplish the bushing repair a portion if not all of the pre-crack was removed when preparing the starting hole. Residual crack lengths for the repaired holes ranged from not visible (zero) to 0.025 inches. The holes were installed with BushLoc bushings and ForceTec retainers to determine the crack growth life of the repaired holes. The initial cracks in the baseline specimens ranged from 0.066 to 0.076 inches. All specimens, baseline and repaired, were tested at the same gross stress. This means that repaired specimens were tested at higher net stresses due to metal removal. The repairs reduced the edge margin from 2.0 to a range of 1.25 to 1.4 depending on the repair bushing diameter. Results of these tests are shown in Figure 11.

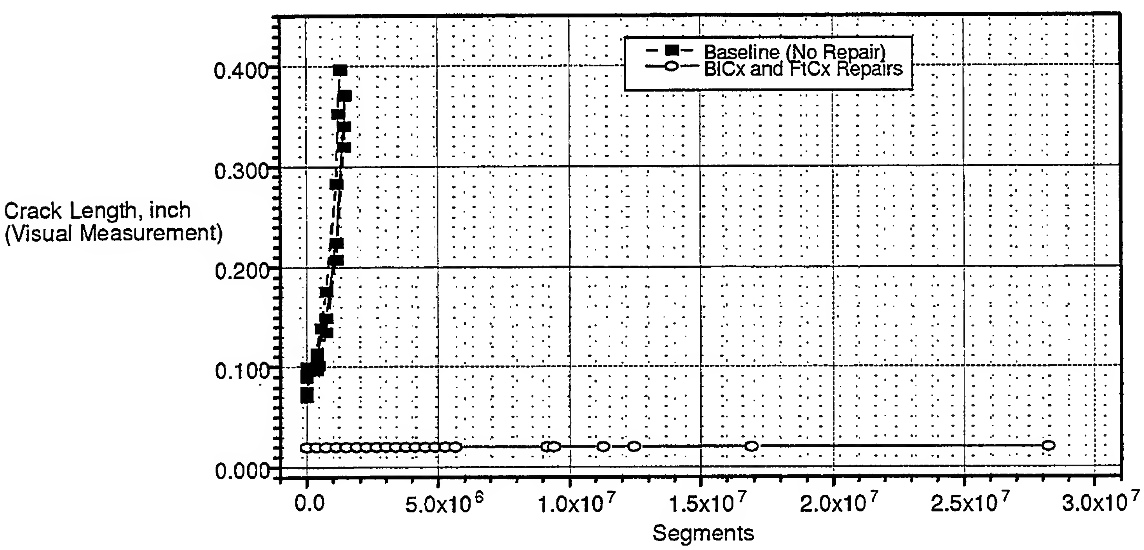


Figure 11: Crack Length Versus Segments

The average flight hours for the baseline specimens was 11,188 hours. All repaired specimens ran to run-out life of 45,405 flight hours with very little additional crack growth. Some of the repaired specimens with the longest flaws were placed back into the test frame in an attempt to produce a failure. These specimens ran an additional 180,000 flight hours without any significant crack growth; greater than 23:1 improvement, shown in Figure 12. Figure 13 shows the life comparison even with remaining cracks after rework.

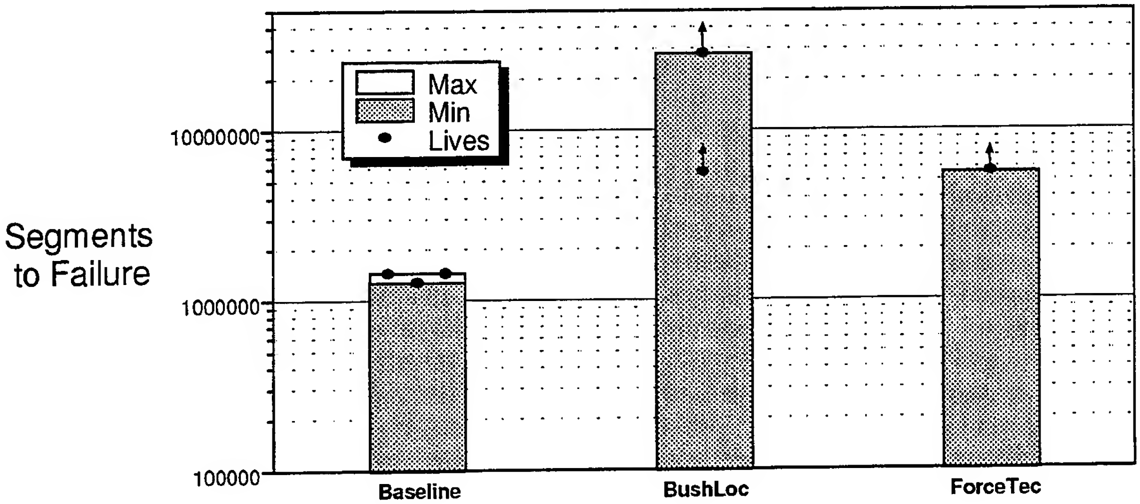


Figure 12: Comparison of Baseline with BushLoc and ForceTec-Repaired Specimens

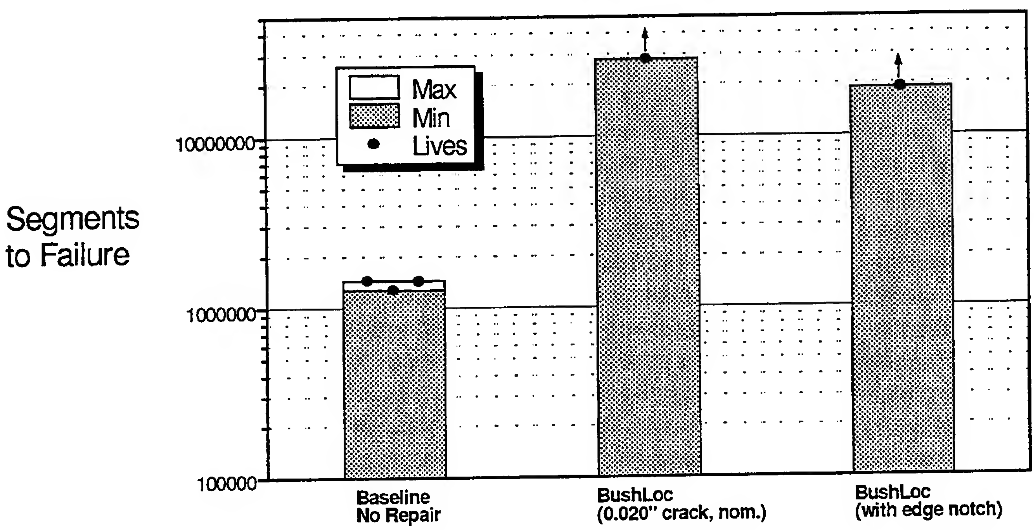


Figure 13: Comparison of Baseline with BushLoc Specimens

This data clearly shows the benefits of using cold expanded bushings for a terminating repair action. The 45,000 additional flight hours provided by the repair is much longer than the remaining service life of these transports. The 180,000 additional flight hours is probably significantly longer than the test life of a new, un-cracked structure.

Additional testing on specimens with a notch at the free edge of the short ligament indicate that the BlCx process did not produce deleterious residual tensile stresses which could induce premature failure. As there was no crack growth at either the notch or the hole, it is unknown whether crack growth would initiate first from the free edge notch and grow towards the hole. FTI believes that crack initiation in this specimen would occur at the hole and not at the edge. Other tests at FTI have been conducted on short edge margin specimens with notches placed at the edge only and at the hole and the edge. Crack initiation in the specimens occurred at the hole.

Commercial Aircraft BushLoc Repair Extensive testing was performed to simulate the repair of a spar cap to wing skin fastener hole on a commercial aircraft. The repair was to remove a large crack from the spar cap without removing the skin or oversizing the original 3/8 inch fastener hole in the wing skin. Several bushing repairs were examined. In one case, a 1/8 inch wall thickness BushLoc bushing was installed. The load transfer test coupon configuration is shown in Figure 14.

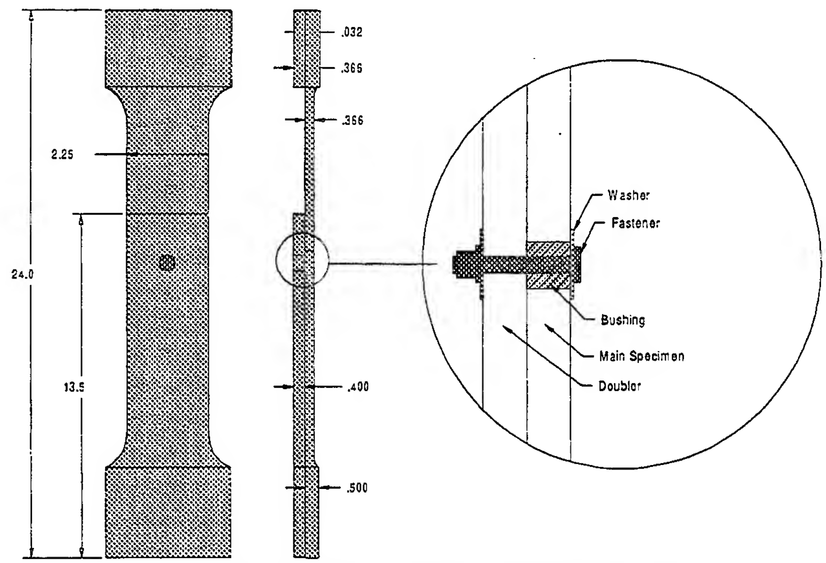


Figure 14: Load Transfer Test Specimen

A typical commercial aircraft wing spectrum load was applied to the specimen and results for various repair scenarios were compared to baseline (fastener only) configured specimens. Test results in Figure 15 show that specimens repaired using either aluminum or steel BushLoc bushings performed better than the baseline configuration and were substantially better than shrink-fit repaired specimens. Results of this BushLoc test were accepted by the FAA as terminating repair actions for this location.

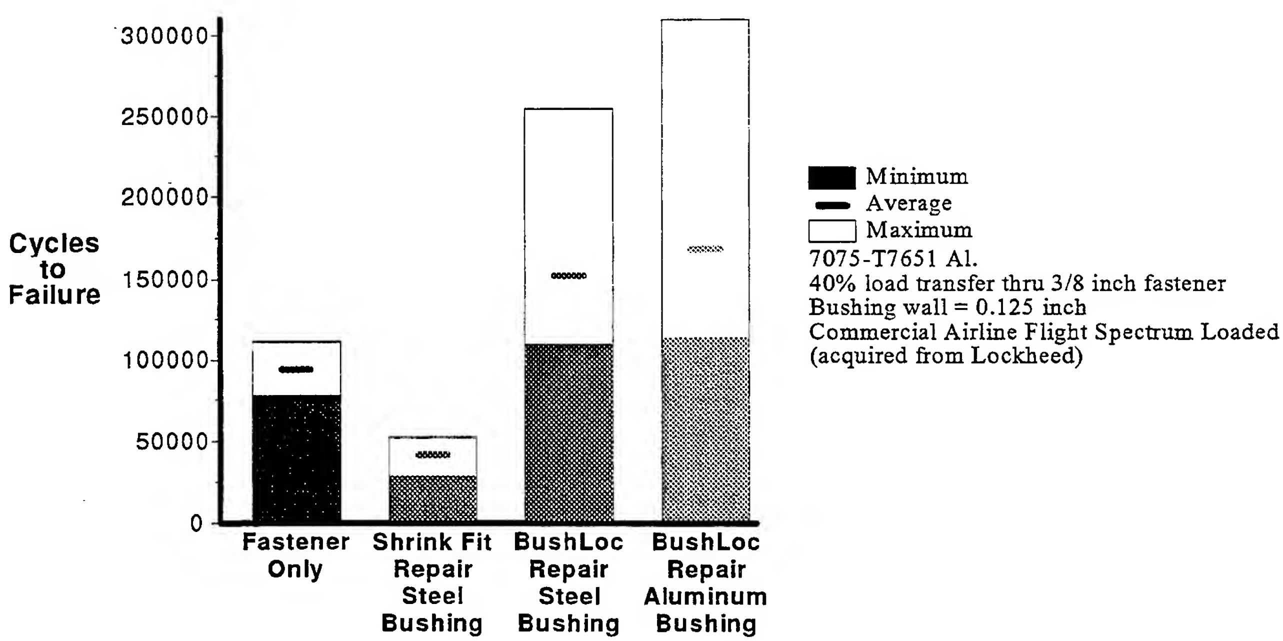


Figure 15: Comparison of Fatigue Life For BushLoc Repaired Specimens

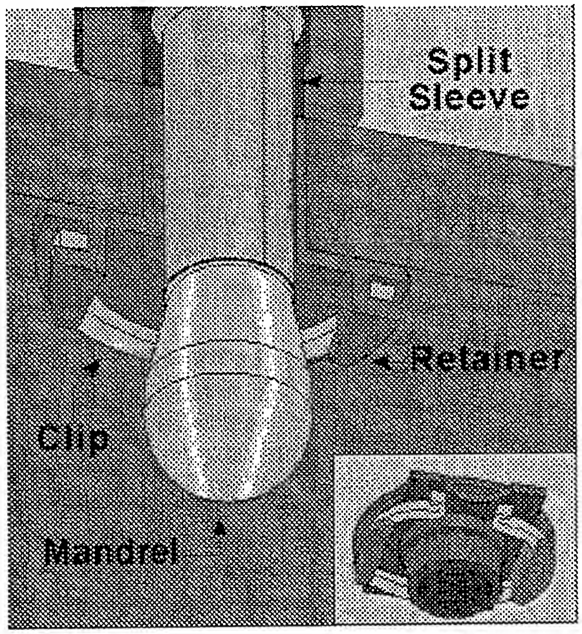


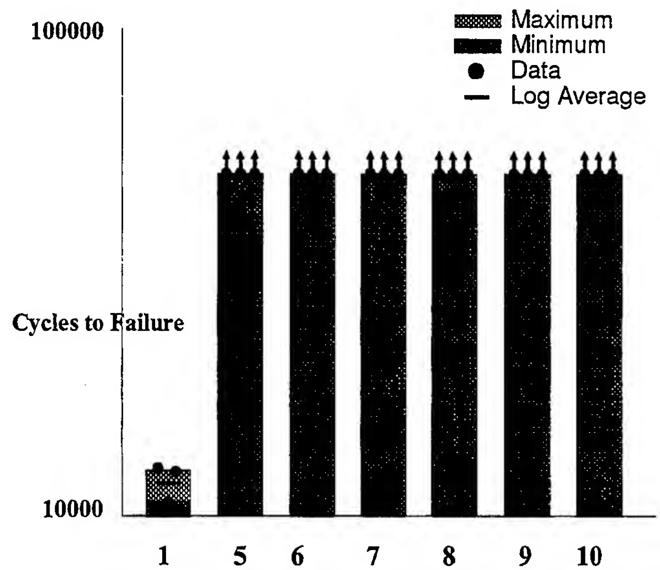
Figure 16: Schematic of ForceTec Rivetless Nut Plate Installation

ForceTec Applications

ForceTec was evaluated along with BushLoc for rework of an F-16 access panel door to repair and prevent cracking from installed riveted nut plates. Zero load transfer specimens were manufactured to simulate the upper fuselage skin of the F-16. The specimens included a fitting with 12 nut plate locations with rivets angled at either 30° or 45° to the load line. Specimens were tested under variable amplitude F-16 spectrum load. Specimens were reworked at 2000 equivalent flight hours (this was the expected time for rework to be carried out on the aircraft) by removing the riveted nut plates, split sleeve cold expanding the satellite rivet holes and plugging them with rivets, and then installing FTI's ForceTec rivetless nut plates into the fastener hole. After rework, testing was continued on the specimens until failure or 16,000 equivalent flight hours. Failure was defined as a crack through the web from the hole to the edge of the specimen. Both of the specimen configurations fitted with ForceTec nut plates and interference fit pins in the rivet holes produced life improvements of at least 3:1 over baseline specimens.

Similar repairs are being carried out on Airbus A300-600 commercial aircraft lower wing access panel holes. In this case riveted gang channels were removed and replaced with individual ForceTec retainers.

The previously mentioned C-141 evaluation also included testing of installed ForceTec retainers to measure the effect of reduced edge margin. Edge margins as low as 1.25 demonstrated greater than 5:1 life improvements. A number of specimens were tested with residual cracks after retainer installation. There was no evidence of further crack growth after testing. Results are shown in Figure 17.



Config	Bushing Type	Flange Loc.	Starting Hole Dia., In.	e/D (e=0.5 in.)	Crack Length, in.
1	Baseline	N/A	0.25	2	0.07
5	ForceTec	Back	0.35	1.4	0.02
6	ForceTec	Back	0.375	1.3	0.008
7	ForceTec	Back	0.4	1.25	0
8	ForceTec	Front	0.35	1.4	0.02
9	ForceTec	Front	0.375	1.3	0.008
10	ForceTec	Front	0.4	1.25	0

Figure 17: ForceTec Repairs in Low Edge Margin Specimens

Review of Military Applications of Cold Expanded Bushings

The following is a summary of the actual and evaluated cold expanded bushing repairs on military aircraft.

- T-38, ForceMate bushing repair for fatigue prevention at lower wing skin holes.
- F-111, ForceMate preventative/terminating repair action on Overwing Longeron and Carry-Through Box. Riveted nut plate hole repair.
- F-15, BushLoc repair of cracked Aircraft Mounted Auxiliary Drive (AMAD) attachment holes.
- F-16, ForceMate repair of nut plate holes providing for new load-carrying fuselage access panels. Tested similar repair using ForceTec.
- AH-1W, ForceMate repair of stub-wing attachment holes.
- C-17, BushLoc repair of production discrepant holes.

Other obvious candidates include:

- JSTARS, BushLoc/ForceTec general repair of fatigue/corrosion damaged holes in these aged B707 structures.
- C-141, BushLoc/ForceTec repair of wing splice fastener holes.
- P-3, BushLoc general repair of corroded/damaged holes.

Repair of Composite Materials

The benefits of cold expanded bushings for repair of damaged holes is not just limited to metal structures. They have been successfully demonstrated for use in composite materials as well. Holes in composite materials can experience damage such as delamination due to drilling, water ingress, fastener insertion and removal, lightning strike, impact, etc. Recent testing of ForceTec retainers in graphite-epoxy material under compressive static loads showed the retainers ability to restore the open hole compression strength of the laminate. The presence of an open hole in a composite can lead to reductions in compressive strengths of up to 35%. By installing cold expanded bushings into holes the laminate strengths were restored to 95% of a no-hole specimens. This data, shown in Figure 18, has significant implications for load bearing structure with clearance fit holes.

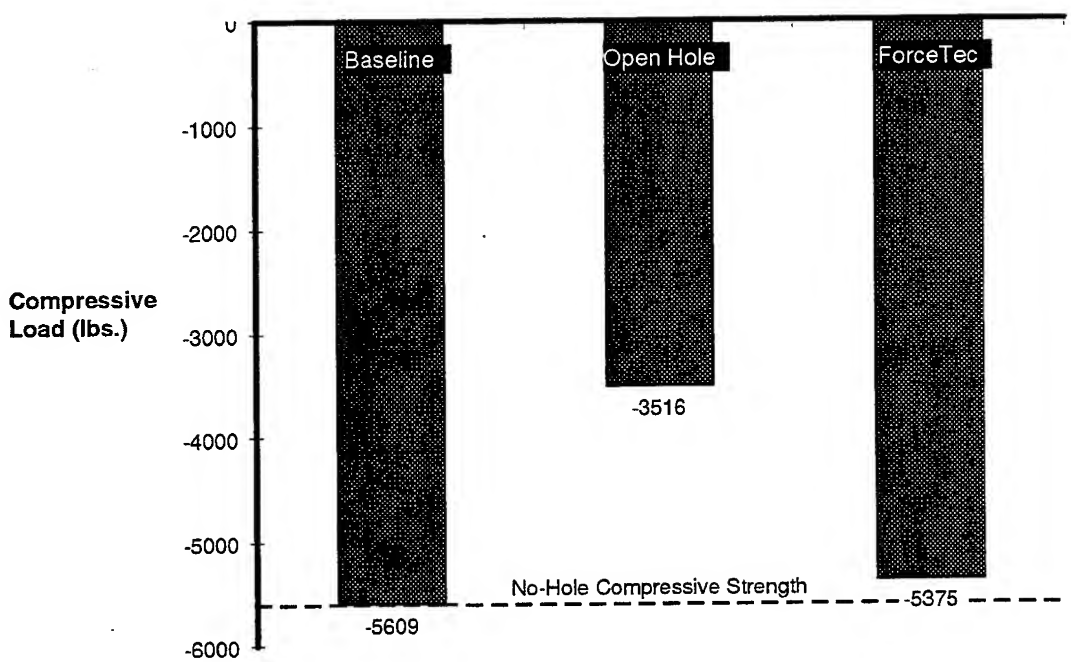


Figure 18: Compressive Strength Comparison for Graphite Epoxy Laminate

Further benefits of cold expanded bushings in composites include lightning strike resistance. Thin wall grommets installed into a graphite epoxy composite showed superior resistance to lightening strike than conventional epoxy installed grommets. The parameter contributing most to the increased lightning strike resistance is believed to be the very tight fit between bushing and hole. Arcing, a primary cause of lightning strike damage, tends to occur across the gaps formed by loose fit bushings or in the air bubbles in conventional bonding agents. The possibilities of using cold expanded bushings in either production or repair of composite structures are exciting.

SUMMARY

When a terminating action is required to repair damaged or discrepant holes, cold expanded bushings should be considered for aircraft in the USAF fleet. The superior benefits of cold expanded bushings in fatigue, damage tolerance, improved corrosion resistance, ease of installation and the flexibility of adapting to almost any repair configuration, gives a technically and economically advanced alternative to traditional repair methods. The cost of using cold expanded bushings is generally similar to the cost of a conventional repairs with the added benefit of providing terminating repair action or at least reduced inspections, and reduced follow-on maintenance costs. The application of cold expanded bushings is simple, safe and efficient and as shown on recent testing on a military airplane programs significantly improve both the fatigue life and damage tolerance of the repair. Testing on one military transport aircraft program included BushLoc and ForceTec with short edge margins, and yet still showed at least a 23:1 fatigue life improvement.

In this era of continually declining defense budgets, it makes sense to save taxpayer money by installing repairs that maximize fatigue life and damage tolerance. Service life extension programs should be designed to get the most fatigue life possible, and the use of Cold Expanded bushings helps to realize this goal. The use of ForceMate, BushLoc and ForceTec Cold Expanded bushing systems to repair damaged and discrepant holes optimizes the use of time, talent and economic resources.

Virtually any aged aircraft is a candidate for these expanded bushing repairs.

Acknowledgment

The authors would like to acknowledge the assistance given by Eric Easterbrook (formerly of Fatigue Technology Inc.) in preparing this paper.

Lasershot Peening of Metals - Techniques and Laser Technology

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Abstract

Laser shot peening, a surface treatment for metals, is shown to induce compressive residual stressed of over 0.040 inch to provide improved resistance to vairous forms of failure. A solid state laser technology employing Nd:glass slabs and phase conjugation enables this process to move into high throughput production processing. Details of the technology are discussed.

Introduction

Various forms of cold working have been used by industry for many years, to induce beneficial compressive stresses in metals. These include fillet rolling, cold expansion of holes, shot peening and the newest form, laser shot peening. The significant increases in resistance to fatigue, fretting, galling and stress corrosion are well known. Shot peening has been the process most widely used because of its ability to induce these stresses efficiently and inexpensively on parts of complex geometry. The depth of the compressive stress produced by shot peening is limited by the kinetic energy transmitted by the force of the shot and the indention of the surface which can reach 0.030 inch but might leave an undersiable surface finish.

Laser shot peening, a surface treatment for metals, employs laser induced shocks to create deep compressive residual stress of over 0.040 inch with magnitudes comparable to shot peening. Laser shot peening is a more damage tolerant process that has generated sufficiently impressive results to move it from a laboratory

demonstration phase into a significant industrial process. However until now this evolution has been slowed because a laser system meeting the average power requirements for a high throughput process has been lacking.

A laser system appropriate for peening at an industrial level requires an average power in the multi-hundred watt to killowatt range and an energy of around 100 J/pulse and pulse duration of 10s of nanoseconds. Pulsed lasers, with output energies exceeding 10 J, have historically been limited to low repetition rates and consequent low average output power. The large fusion lasers such as Livermore's Nova Laser and the University of Rochester's Omega laser can produce single pulse energies at 1 micron wavelength in the 100 kJ range but are limited to firing about once every two hours for an equivalent average power of only tens of watts. Commercially available lasers, with outputs of 10 to 100 J, if available at all, are limited to repetition rates around 0.25 Hz, an average power of 25 W. In this paper we report on a highly developed laser technology employing Nd:glass slabs and a master oscillator/power amplifier with wavefront correction called phase conjugation which for the first time pushes the average power output into the 500 W to 1 kW range and meets the requirements for industrial laser peening.

Laser Shock Peening

With the invention of the laser, it was rapidly recognized that the intense shocks required for peening could be achieved by means of tamped plasmas which were generated at metal surfaces by means of high energy density (~200 J/ cm^2), lasers with pulselengths in the tens of nanoseconds range. Initial studies on laser shock processing of materials were done at the Battelle Institute (Columbus, OH) from about 1968 to 1981[1,2]. Excellent recent work has also been reported in France[3]. Figure 1 shows a typical setup for laser peening. Laser intensities of 100 J/cm^2 to 300 J/cm^2 in a pulse duration of about 30 ns can generate shock pressures of 10^4 to 10^5 atmospheres when absorbed on a metal surface (a thin layer of black paint on the surface provides an excellent absorber) and inertially confined with a surface layer (tamp) such as water. These shocks have been shown to impart compressive stresses, deeper than those achievable with standard shot peening. Special techniques in controlling the pulse temporal and spatial shape are used to prevent the high intensity laser from breaking down the water column or generating stimulated processes which reflect the laser energy before reaching the paint surface.

With appropriate care given to the setup, impressive results can be achieved from laser peening.

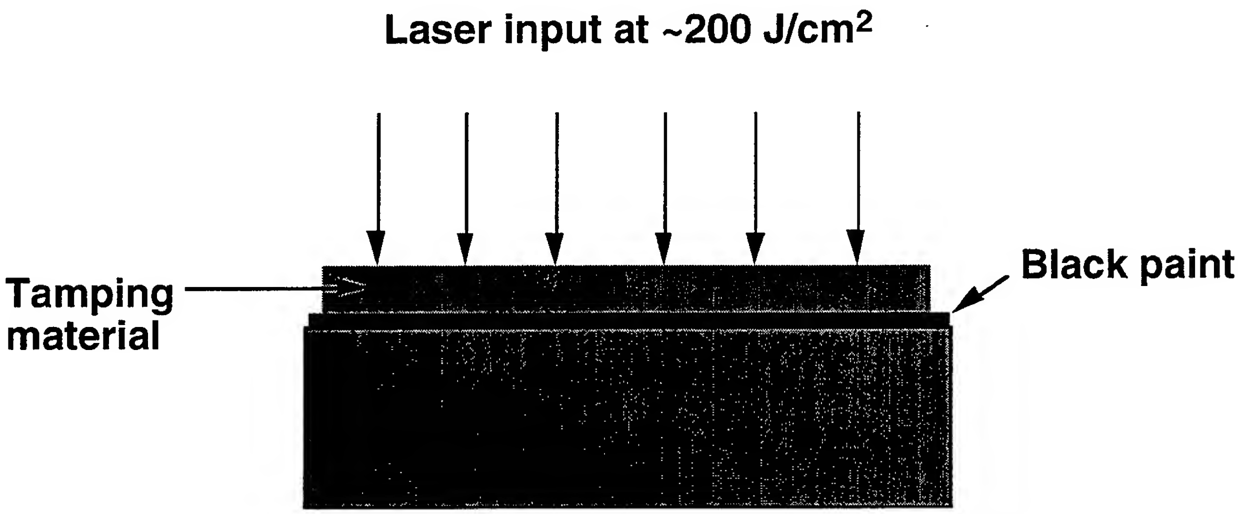


Figure 1. Typical setup for laser peening including a input laser beam of 200 J/cm2 and 30 ns pulselength. The metal layer is covered with a layer of paint to provide light absorption and is covered with a thin water tamping layer to contain the shock.

AS an example of the laser process, Figure 2 shows the residual stress induced in Inconnel (Ti-6Al-4V) by laser peening and contrasts it with typical results achieved by shot peening. Clearly the laser generated shock can be tailored to penetrate deeper into the material and create a significantly greater stress volume. Induced residual stress prevents treated parts from developing cracks due to stress corrosion. Additionally other types of corrision will require longer periods of time to penetrate the compression layer induced by Lasershot Peening. Deep residual stress is important for critical areas of components such as turbine blades because it prevents debris damage from penetrating beneath the compressive layer. Foreign object debris (FOD) picked up in operation can often generate damage sites which penetrate a thinner compressive layer and hence become an initiation point for fatigue cracks.

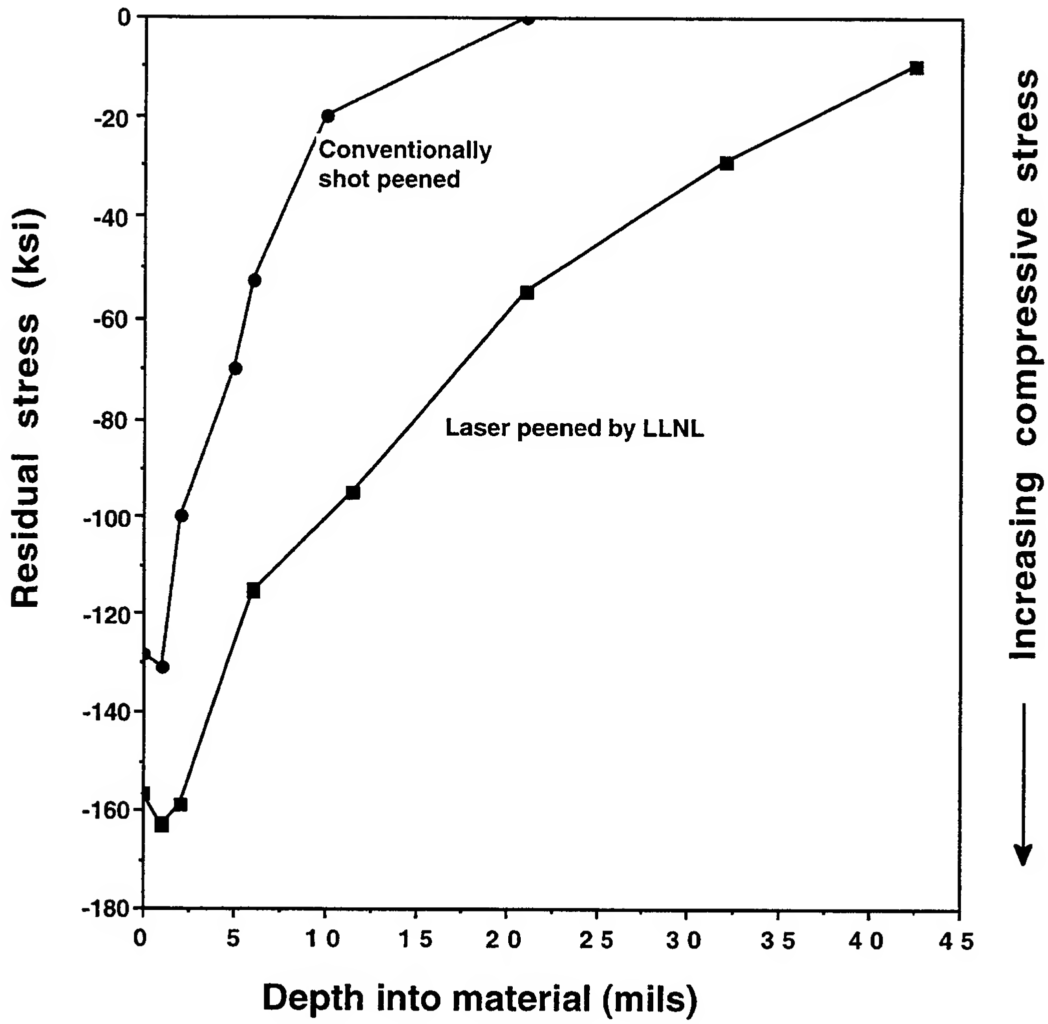


Figure 2. Residual stress induced by laser peening can be made deeper and with significantly greater stressed volume than conventional shot peening.

Another important element in obtaining deep residual stresses is the use of successive shocks to drive the stress deeper and deeper while not exceeding materials limits at the surface. Figure 3 shows results of successive applications to a titanium surface (Ti-6Al-4V) of laser pulses at 200 J/cm^2 and pulse duration of 30 ns. As can be seen, the application of a first and then a second shock successively drives stress deeper and into the material.

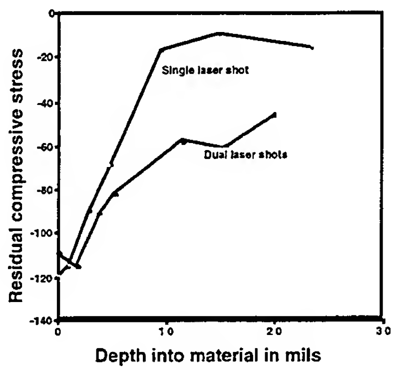


Figure 3. A first and then second lasershot pulse can generate successively deeper residual stress and thus larger stress volume.

In testing on operational components, such as jet engine fan blades, researchers have shown this treatment to be superior for strengthening new and previously damaged fan blades from fatigue and corrosion failure[4]. However the laser technology for doing these types of tests has been limited to producing pulses less than once per second thus peening areas of about 1 square centimeter per second. This rate is acceptable for laboratory demonstrations but clearly not meaningful for cost-effective production. Since the cost of any high energy (100 J) laser is dominated by the hardware required to achieve the single pulse energy, it is imperative to have high repetition rate capability (~10 Hz) in order to keep the production cost per laser shot low.

High power laser technology at LLNL

The Laser Programs Directorate at Lawrence Livermore National Laboratory (LLNL) has been a world leader in developing high energy Nd:glass lasers for fusion applications for the past 25 years. The Nova laser, producing over 120 kJ per pulse, routinely fires 8 to 10 shots per day for dedicated fusion and nuclear effects studies. More recently, the Livermore Laboratory has been directed by the Department of Energy to proceed with building a newer facility, the National Ignition Facility (NIF) which will produce over 2 MJ per pulse of energy in one or several shots per day and is intended to produce more fusion energy release than laser energy input. It is

clear that enormous successful investment has been made to develop high energy solid state lasers.

Generating high energy from a solid state glass laser is not a significant issue. However generating high average power at high energy has been. Over the past decade, LLNL has been developing higher average power systems with energies (depending on the application requirements) of 25 J to 100 J/pulse. This laser technology now has repetition rates of up to 10 Hz and average powers near 1 kW. The technology development has been supported by the Advanced Research Projects Agency (ARPA) of the Department of Defense and more recently by the US Navy and US Air Force. The ARPA funding was focused on converting the infrared light to high average power x-rays (10 Å). This short wavelength has interest as a light source for proximity printing of advanced generation integrated circuits. The Navy and Air Force funding was directed toward obtaining a light source for long range and highly coherent illumination of missiles and space objects. One of our LLNL lasers is currently in service at a Navy facility at the Kennedy Space Center, Cape Canaveral Florida and a second more power unit is being delivered to the Air Force Phillips Laboratory, Albuquerque New Mexico. This technology has allowed us to develop a glass laser system with energy of 100 J/pulse, adjustable pulse length from 10 ns to 1 μs, near diffraction limited beam quality and average power up to 600 W. This laser technology is ideal for the laser peening application and by a factor of 20 to 50 exceeds the average power achieveable by any commercially available laser technology.

The High Average Power Nd:glass Slab Laser System

A system suitable for laser peening must output an energy in the range of 25 J to 100 J per pulse. The throughput of a peening system will then highly depend on the average pulse repetition rate that the laser can achieve. A laser system based on Nd doped glass gain media is the only identified technology that can realistically achieve this type of energy output with acceptable pulselength. Such a system is typically based on an oscillator and one or more rod amplifiers which are optically pumped by flashlamps. As an unavoidable consequence of providing the optical gain, the flashlamps deposit heat into the glass. This heat must be removed at a rate commensurate with the rate of deposition, that is, the pulse rate of the laser. Thus

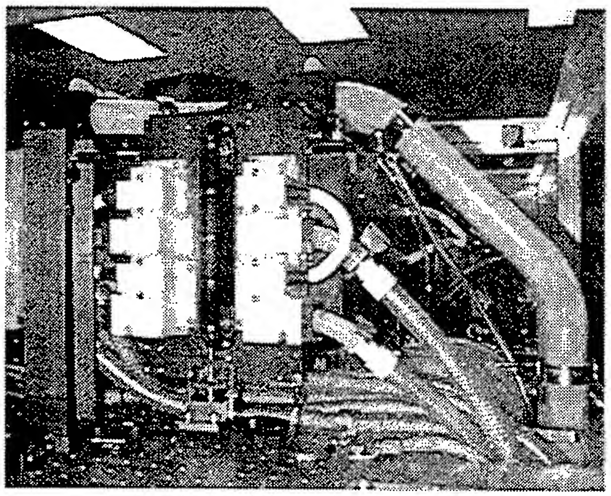
the glass must be cooled, typically by flowing water. As the glass is simultaneously heated and cooled a thermal gradient develops from the center to edge of the glass. This gradient stresses the glass, inducing wavefront deformation and very significantly depolarization of the beam. Thus the thermal loading of the laser gain media is a major limitation to the available average power that can be extracted from the laser. As the repetition rate of the laser is increased, the thermal loading correspondingly increases and degenerates the laser performance often depolarizing and aberrating the laser beam to the point where the laser optics damage. In the limit, the loading will fracture the glass. The LLNL laser design alleviates this thermal problem in three ways; 1) the slab gain medium is pumped in a highly uniform manner minimizing depolarization and distortion, 2) the laser beam is propagated through the slab in a zig-zag manner to average out much of the wavefront distortion and 3) SBS phase conjugation highly corrects residual wavefront distortions.

The LLNL Nd:glass slab laser system

The LLNL high average power Nd:glass laser technology is comprised of a single master oscillator and one or more power amplifiers. The amplifier gain medium is neodymium (Nd) doped phosphate glass APG1 supplied by Schott Glass Technologies Inc. or HAP4 supplied by Hoya Corporation. The glass is configured in a slab shape to allow one thin dimension for rapid heat removal. Typical slab dimensions are 1 cm x 14 cm by 40 cm. Typical Nd doping level is 3×10^{20} cm^{-3} or 2.7% by weight.. Figure 4 shows a cross-sectional view of an amplifier. Unlike a more traditional amplifier where the beam is propagated through the gain medium in a straight line, our design employs a zig-zag path path, reflecting the beam internally off the slab faces. As shown in the figure, the slab is positioned in the center of the assembly and has a water cooling channel along both sides formed by the slab face and a reflector window. Two flashlamps on each side pump the slab through the cooling channels. A diffuse reflector surrounds the flashlamps and by appropriate shaping provides uniform optical pumping. The reflector material is made of a Teflon-like substance called Spectralon machined to a specific shape to tailor the pumping irradiance on the slab surfaces. Designing a thin dimension for the gain medium creates one short path for high heat conduction from the slab

center to the cooling water. The resulting high heat transfer efficiently removes the heat buildup and directly increases the repetition rate capability of the laser. Very uniform optical pumping from the reflector assembly results in uniform energy distribution from top to bottom in the slab. At high repetition rates a large

Slab amplifier operating in the AIT laser system



Schematic of the amplifier design

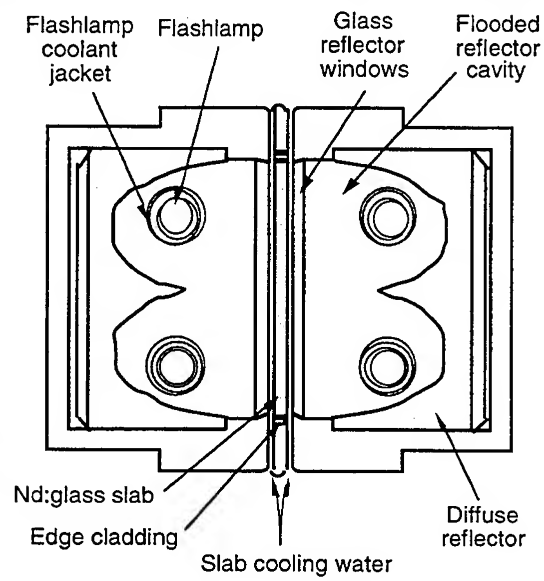


Figure 4. Photograph and cross-sectional view of the Nd:glass laser amplifier. Flashlamp light, tailored for highly uniform illumination by the diffuse reflectors, provides the excitation to the slab. The thin dimension of the slab allows for efficient heat extraction into the water flow. The laser light zig-zags through the slab averaging wavefront distortions.

thermal gradient develops in the slab from center to edge. However, the laser light is propagated through slab so that the beam propagates in a side-to-side zig-zag manner. This zig-zaging averages the side-to-side thermally induced pathlength differences providing a high quality horizontal wavefront even though there is a significant gradient in this direction.

A representative optical layout for the laser system is shown in figure 5. The output of the oscillator transmits through a Faraday isolator and then in P polarization transmits through a polarizing beam splitter. The beam next passes through an input/output Faraday rotator which acts as a passive cavity switch and

then still in P polarization transits through the first polarizer within the amplifier cavity. It next transits a relay telescope, double passes the amplifier and then propagates back through the telescope and through a 90 degree rotator. The now S polarized beam reflects off the polarizing beam splitters, transits through the relay telescope and for a second time double passes the amplifier. Passing again through the 90 degree rotator and converting back to P polarization, the beam now propagates to the SBS phase conjugator. Within the phase conjugator the beam generates an index grating by means of stimulated Brillouin scattering. This index grating acts as a special kind of mirror, reflecting the beam back but with the phase of the wavefront reversed. The beam retraces it path through the amplifier ring and back to the input/output Faraday rotator. In the output direction the rotator flips the beam to S polarization, allowing it to exit the laser. In these final four passes, the beam amplifies to the desired output energy and very importantly, all phase errors accumulated in the first four amplifier passes are negated in the final four amplifier passes (by summation of phases which are basically identical but reversed in sign) to generate a high power beam with nearly diffraction limited beam quality. By correcting for thermal aberrations our design allows us to extract average powers up to the mechanical limit of the gain medium.

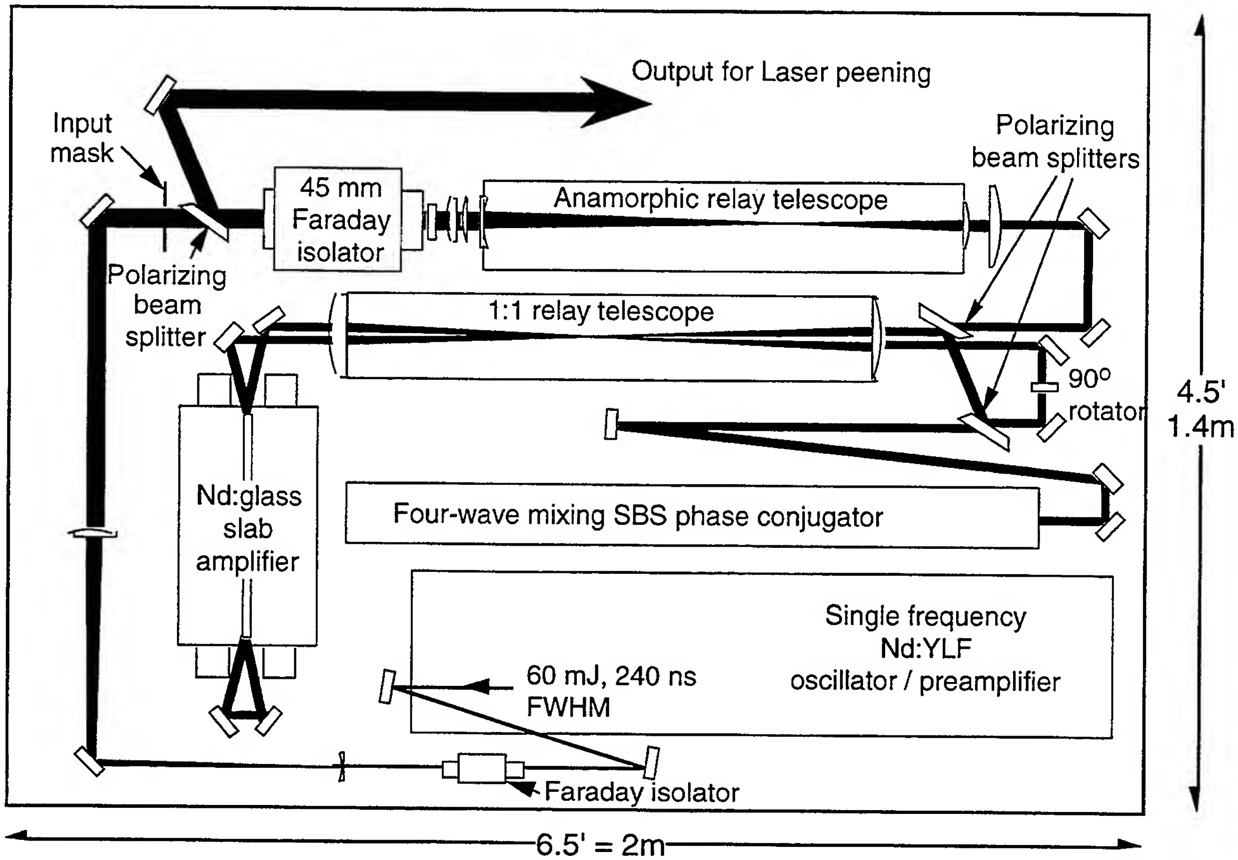


Figure 5. Typical layout of the high energy Nd:glass laser system with repetition rate of 6 to 12 Hz. The high quality master oscillator output is amplifier by means of 8 passes through the slab amplifier. The SBS phase conjugator provides necessary wavefront correction so that the output is near diffraction limited beam quality.

There are additional significant advantages to the operation of the amplifier system with the SBS phase conjugator. Eight gain passes through the zig-zag slab amplifier can be achieved using passive polarization switching in the regenerative amplifier ring. The fact that the SBS cell provides interstage gain isolation makes this possible since, if it were replaced with a mirror, the small signal gain through eight consecutive gain passes would lead to parasitic oscillation from the small reflective lossses of AR coated optical surfaces in the ring or in the output beam.

The SBS phase conjugator also very effectively conjugates the first order aberration of tilt. This greatly reduces the sensitivity of the system performance to small changes of optical alignment in the ring. No change in output power or pointing direction during operation are observed for large mirror misalignments in

the ring. Loss in performance is limited to only to those angular excursions that result in vignetting of the beams at the edges of the amplifier slab.

Finally and specifically for the laser shock peening application, the SBS phase conjugation naturally produces a fast rising edge laser pulse. Because the SBS is a non-linear process with a definite threshold, the phase conjugator does not respond to the initial low intensity buildup typically associated with a laser pulse. The beam returned by the conjugator has its leading edge "clipped" and thus the returned pulseshape has a sharp, sub-nanosecond rising edge. The fast rising pulse is critically important for laser peening because it reduces the possibility of breakdown or other non-linear processing occurring in the tamping material and allows the full pulse energy to reach the paint area on the metal and thus contribute to building the high intensity shock.

Although not needed for shock peening, the laser's high output beam quality enables efficient conversion of the infrared output light into green light. Conversion efficiencies for this laser technology range from 65% at 1 μs duration pulses to over 80% for 10 ns pulses.

Achieving high output energy from a solid state laser is limited by the physical size of the gain medium, the saturation fluence of the material and the damage fluence that can be accommodated. Increasing the height of the slab becomes impractical due to the cost of large optics. Increasing the length effects the gain and amplified spontaneous emission limits. Increasing the thickness directly decreases the average power capability. However using our newly demonstrated technique of phase locking multiple apertures we can scale the laser output into exciting new levels of energy and average power[5]. In this technique a single laser oscillator feeds multiple laser amplifiers and the beams are recombined into a single phase conjugator which effectively locks the separate channels into a single coherent laser beam. The far field beam quality is near the physical diffraction limit, the laser energy is that of the combined multiple apertures and the repetition rate is the higher one associated with a single laser slab. Figure 6 shows a highly engineered 100 J/pulse glass slab laser system packaged and ready for deliver to the US Air Force Phillips Laboratory. This laser will be used in a long (600 ns) pulse mode for space object imaging. Its output energy of 100 J/pulse and its short average power capability of 600 W offers an ideas source for lasershot peening.

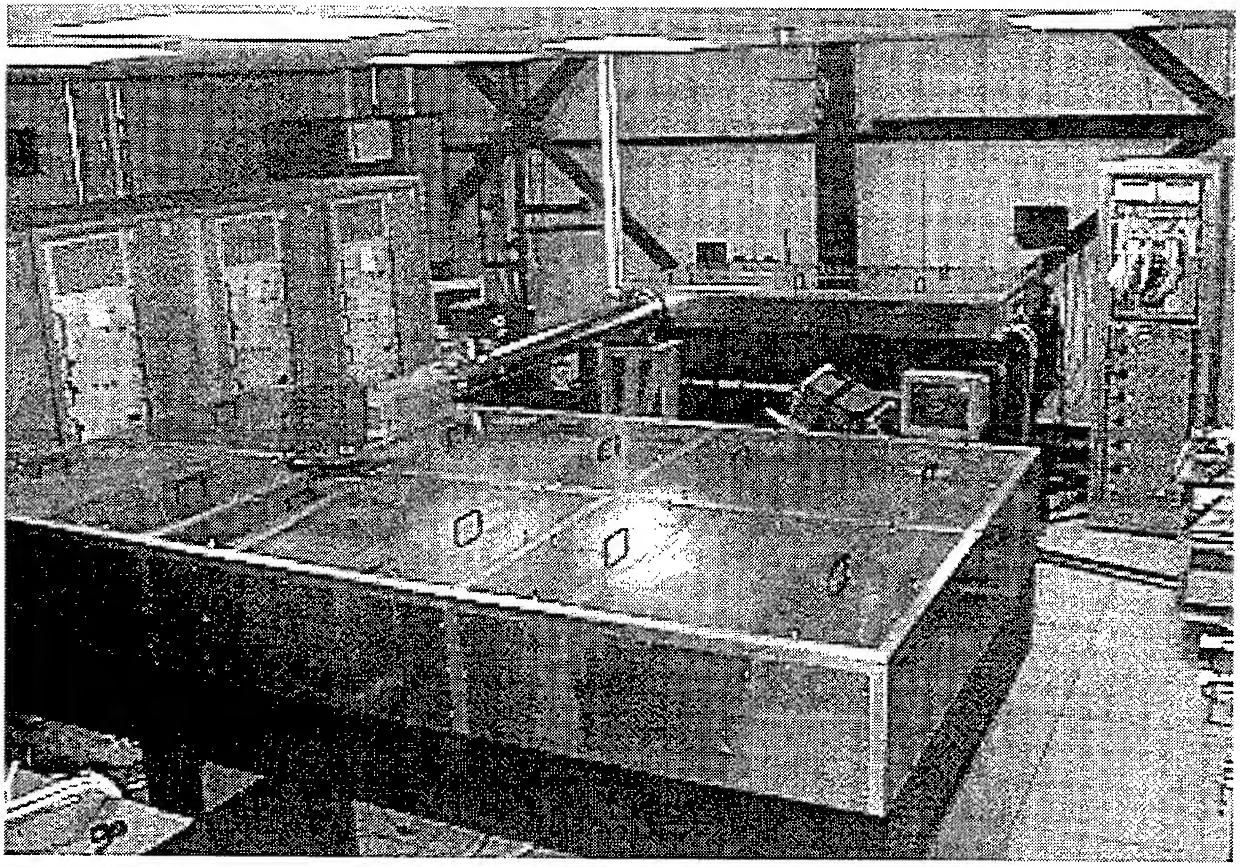


Figure 6. Completed laser, operational and ready for delivery to the US Air Force's Phillips Laboratory.

High Throughput for Laser Peening

The high average power available from the Nd:glass slab laser system enables for the first time a high throughput laser peening system. Assuming that 200 J/cm^2 is required to generate an effective 10 kBar shock, the LLNL laser system, operating at 100 J per pulse and and 6 Hz repetition rate, will have a throughput capability in excess of 10,000 cm^2 per hour for single pulse applications and 5,000 cm^2 for dual pulse applications. Upgrading the laser with the newly developed APG-2 laser glass and thus doubling its average power output, the throughputs can be increased to 20,000 cm^2 per hour and 10,000 cm^2 per hour, respectively.

Summary

We have developed a class of laser system at the 100 J level with average power capability pushing toward 1kW. Its new technology includes uniformly pumped zig-zag slab gain media, master oscillator/power amplifier (MOPA) architectures and phase conjugation, to minimize the effects of thermal loading and correct the problems generated by it. This technology enables for the first time industrial application of a high throughput laser peening process.

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Development of Laser Shock Peening of Airfoil Leading Edges for Single Engine Weapon Systems

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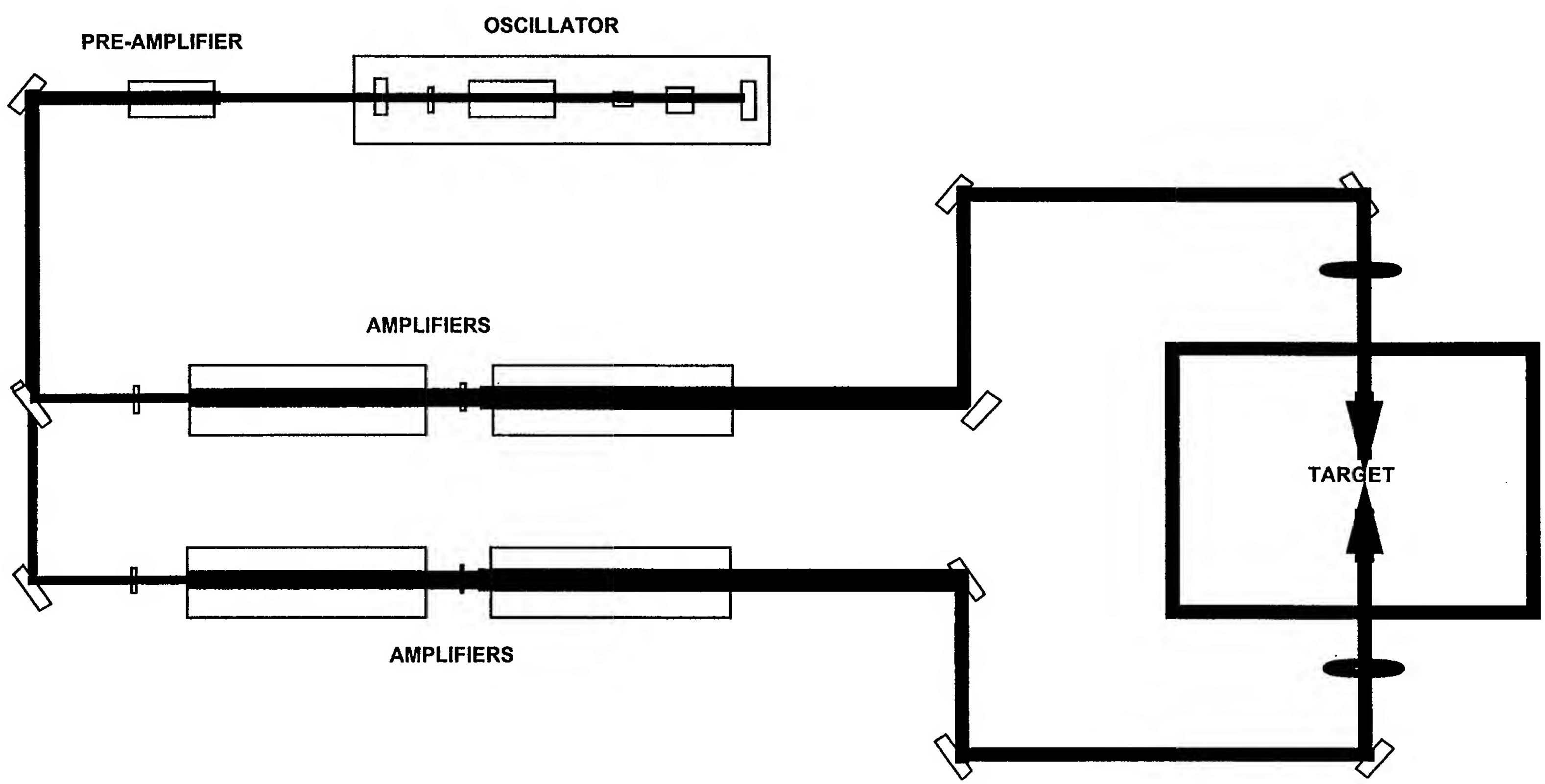
General Electric Aircraft Engines Cincinnati, Ohio

OUTLINE

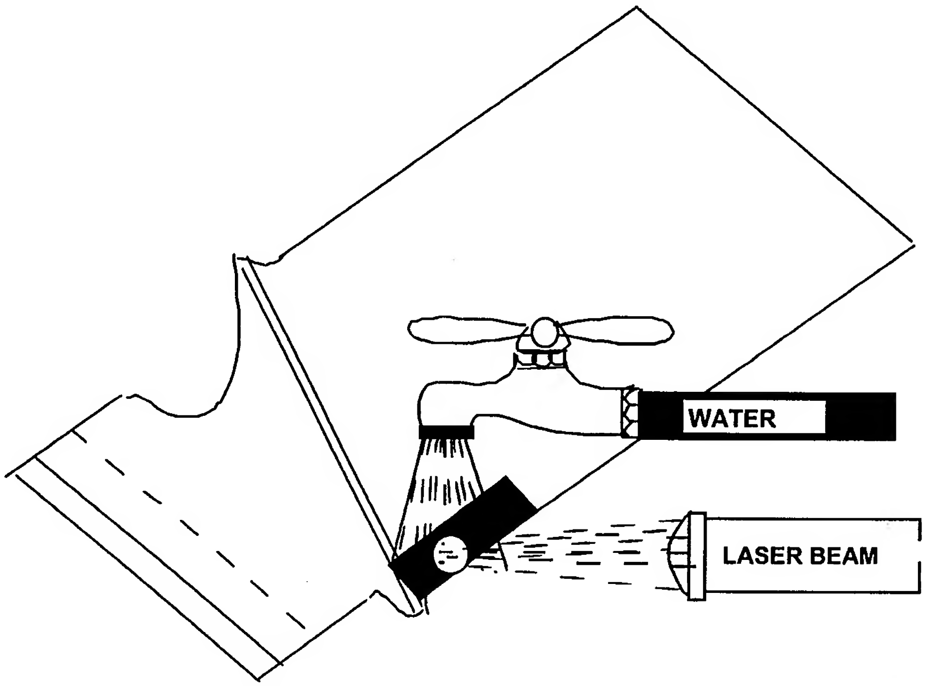
- LSP Laser System for Airfoils
- LSP Concept
- Advantages of LSP
- Analytical LSP Residual Stress Distribution
- Testing
- Conclusion

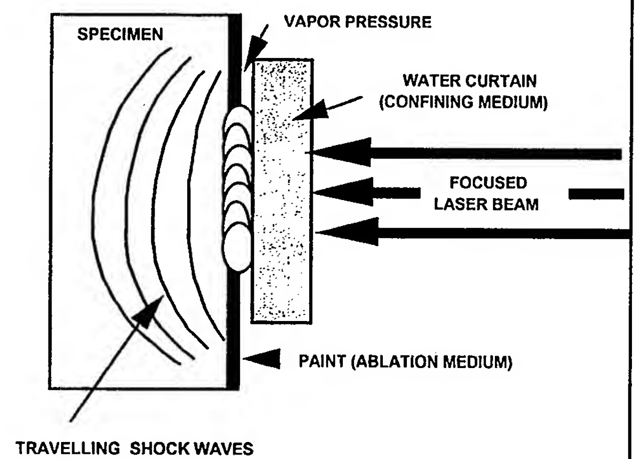
Typical LSP System for Airfoils



ASIP 1997 LSP W. Cowie, S.R. Mannava, T. Compton: GEAE

LSP CONCEPT





Residual stresses are generated by traveling shock waves created by rapidly expanding vapors due to absorption of high energy laser pulses. Paint protects surface and water directs shock waves into the material.

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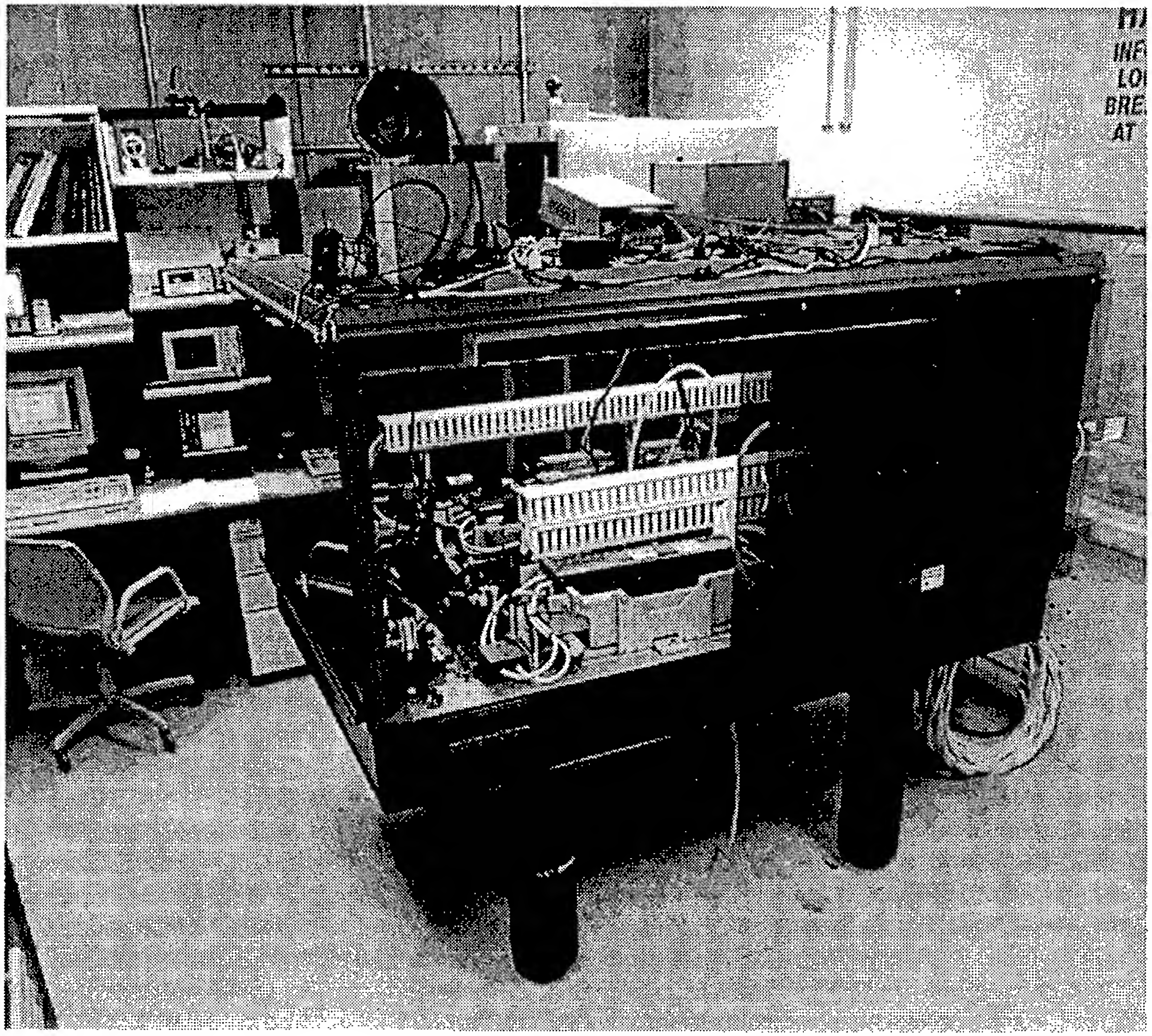
LSP LASER SYSTEM



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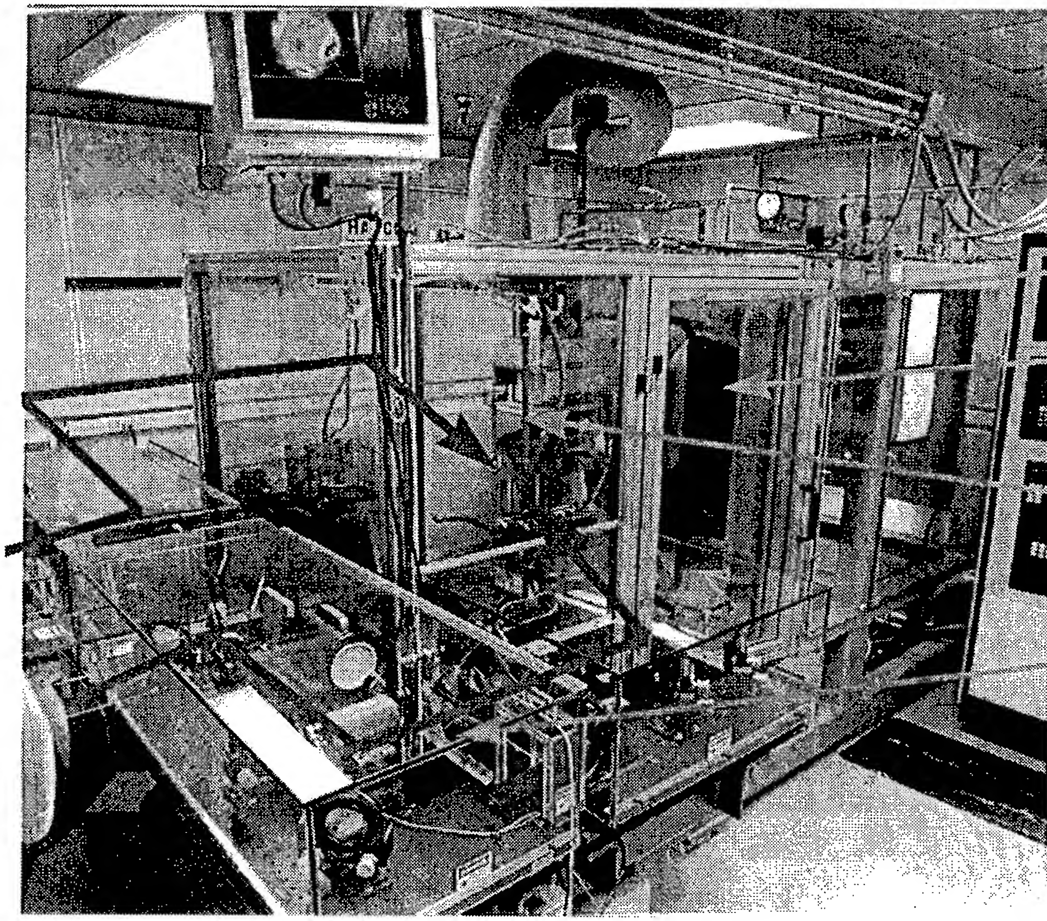
LSP BEAM DELIVERY & PROCESSING CHAMBER



Process Chamber

Part Manipulator

Water Nozzles

Beam Delivery

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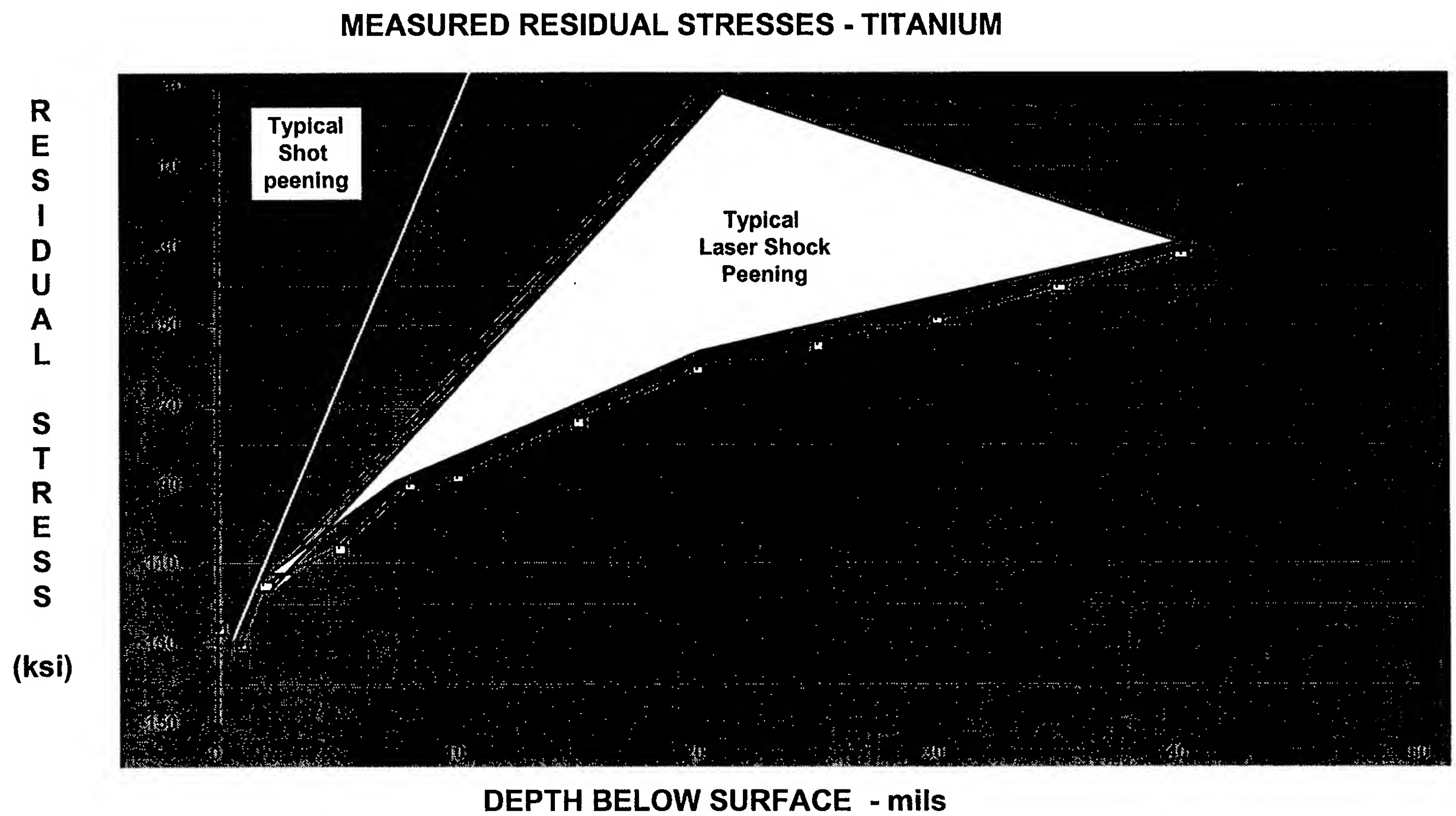
Advantages of LSP



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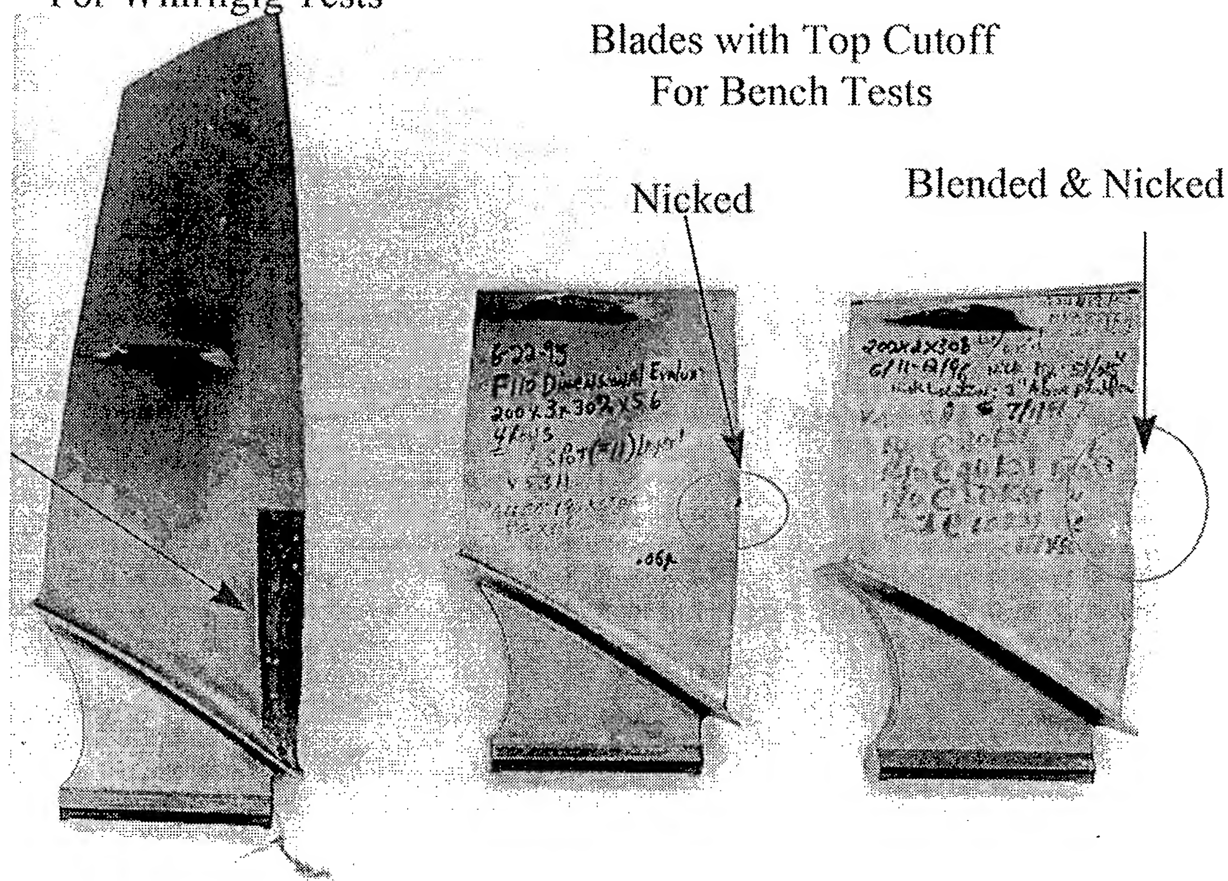
F110 TEST BLADES

Full Size Blade
For Whirligig Tests

Blades with Top Cutoff
For Bench Tests



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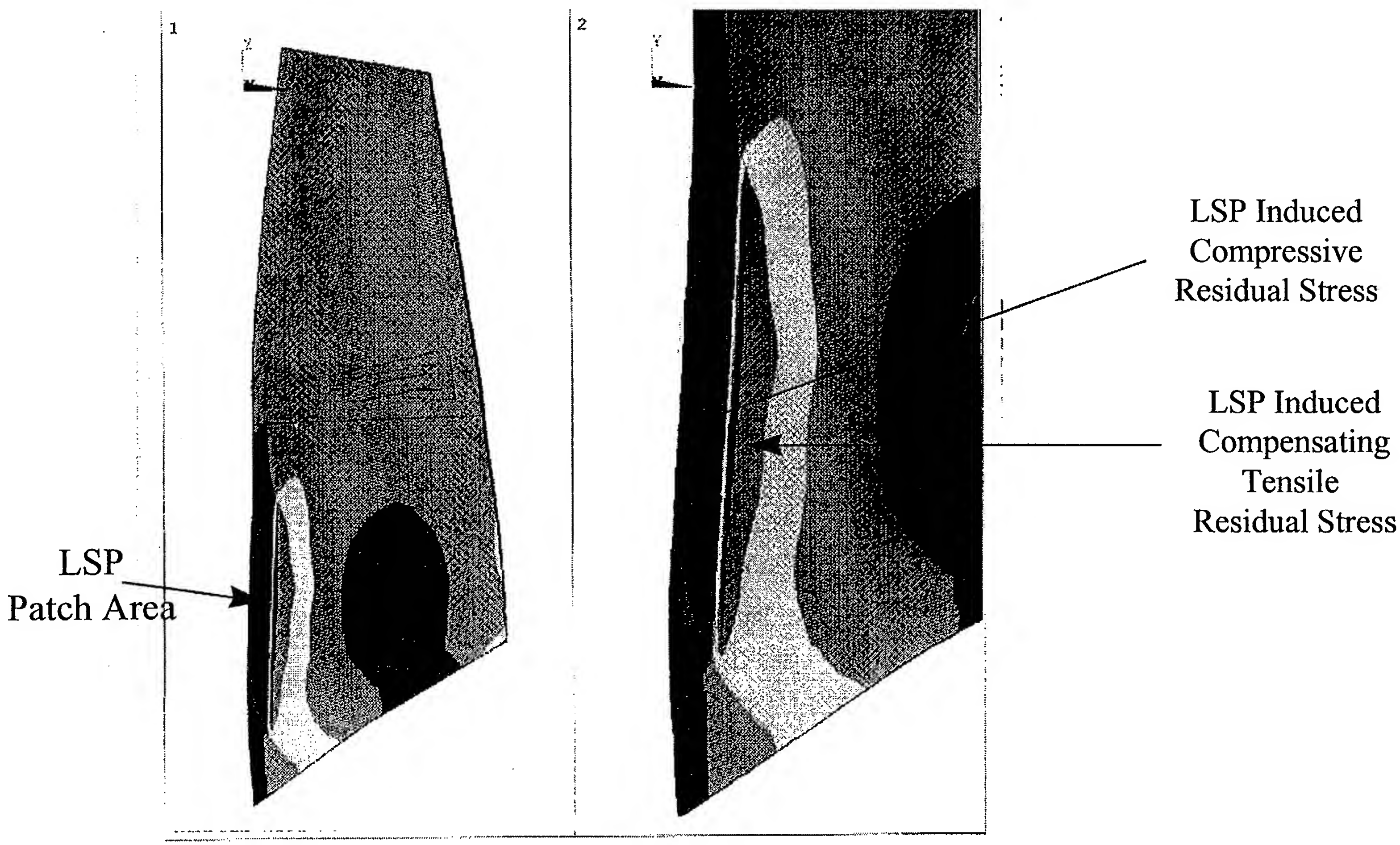
TYPICAL CALCULATED LSP RESIDUAL STRESS DISTRIBUTION



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Component HCF Test Results

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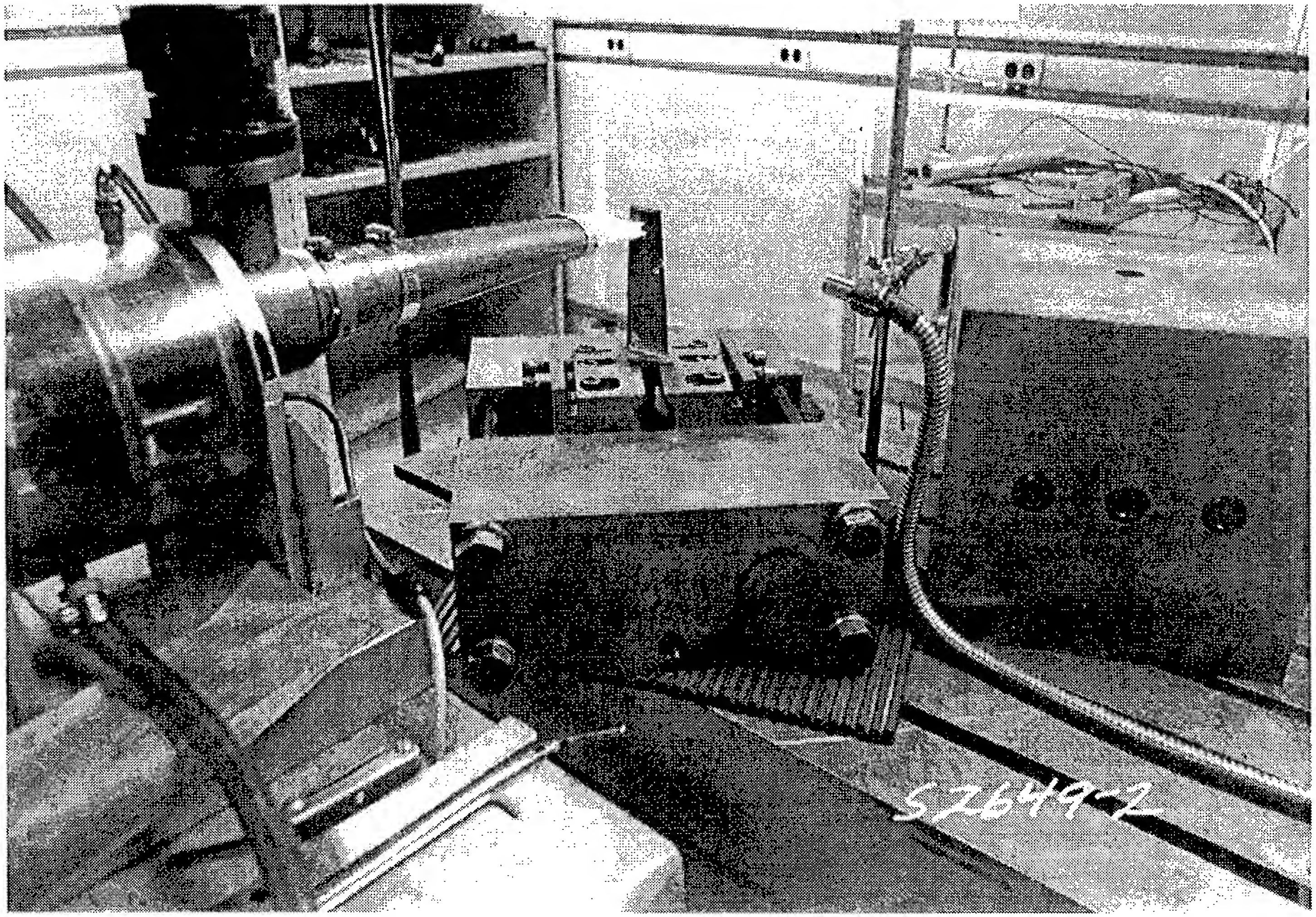
SIREN HCF TEST FACILITY



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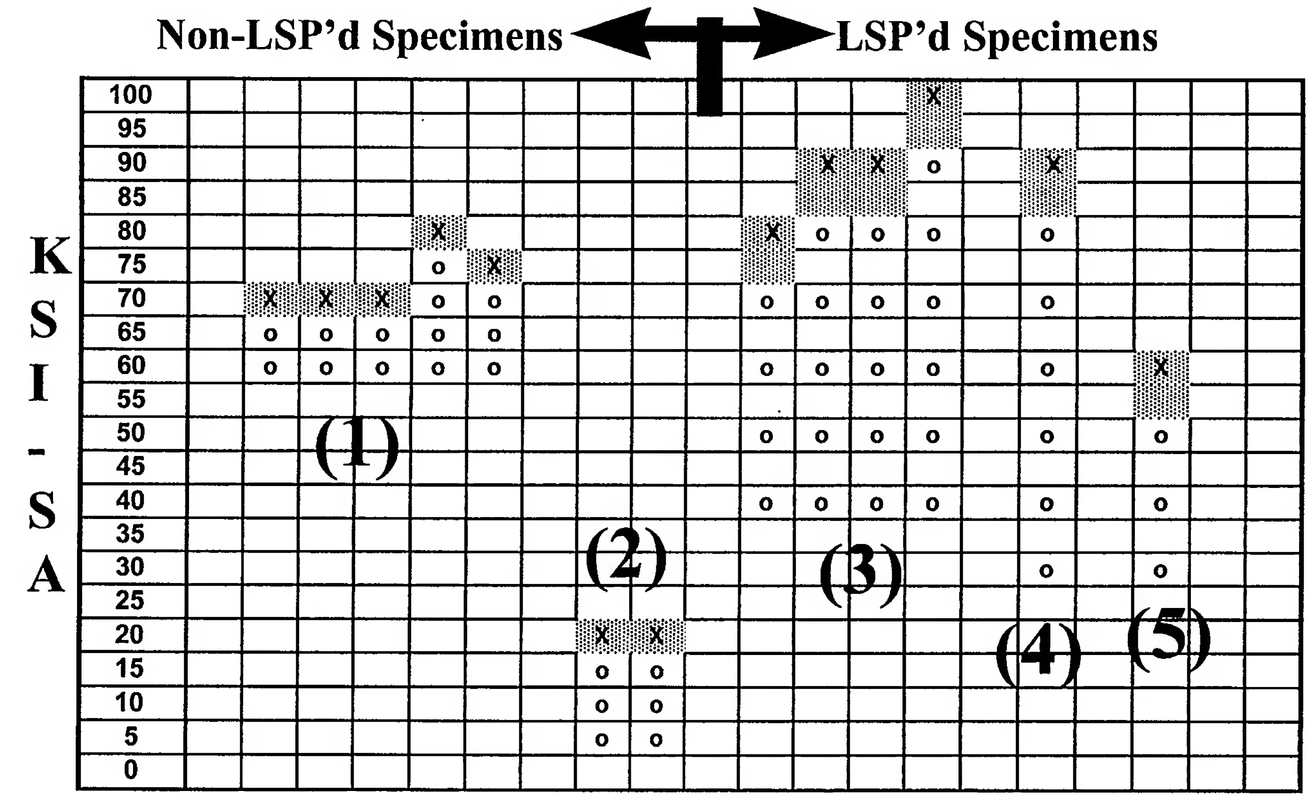
HIGH CYCLE FATIGUE TEST DATA

O- Runout 10^6 Cycles X- Failure



(1): Baseline Data, UnDamaged, Non- LSP Treated
(2): Baseline Data, Damaged, Non- LSP Treated
(3): LSP Effect Data, Damaged, LSP Treated
(4): AMT Engine Blades, Damaged, LSP Treated
(5): AMT Engine Blades, Blended (0.15"), Damaged, LSP treated

LSP RESTORED HCF STRENGTH CAPABILITY

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WHIRLIGIG RIG TEST

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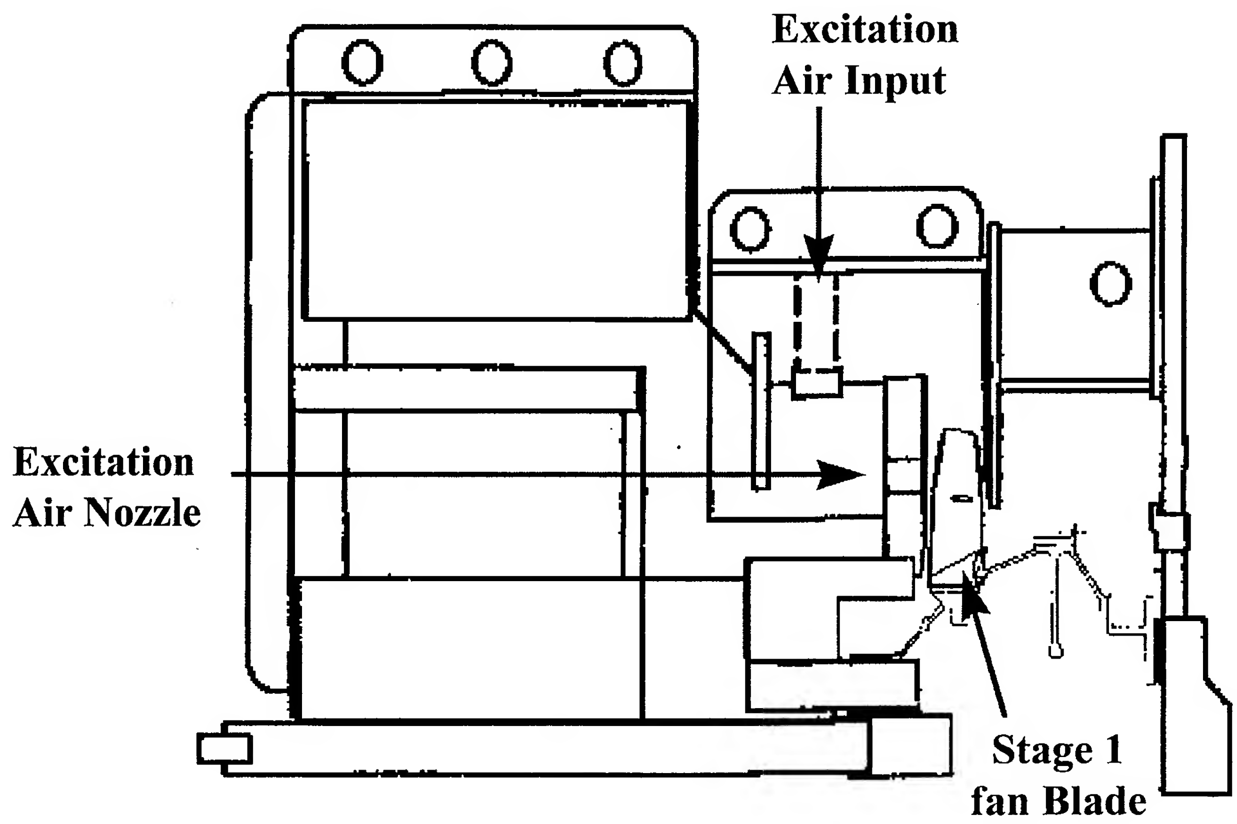
WHIRLIGIG TEST SETUP



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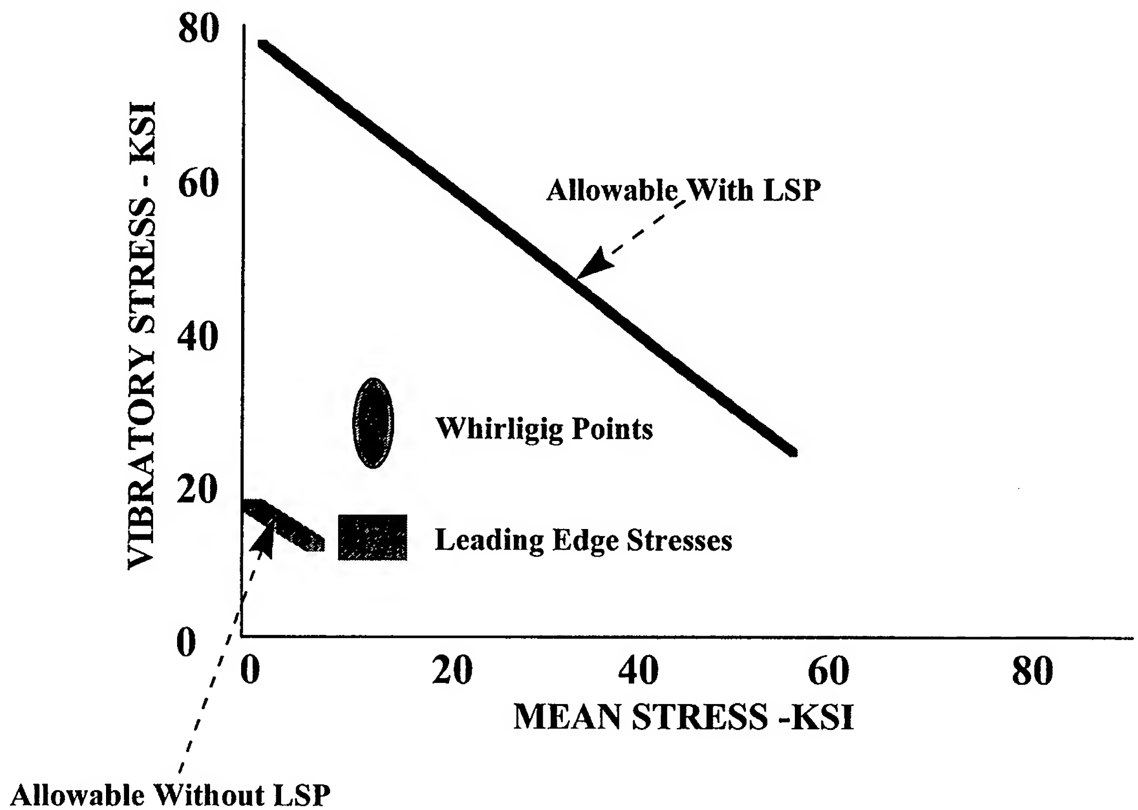
WHIRLIGIG TEST RESULTS

- Three LSP treated blades with Sharp "V" Notches 0.020" - 0.025" Deep
  - Strain Gage Adjacent to Notches Plus Engine Correlation Point
- Approximate Temperature in Rig is 300-350 F
  - Same as Highest Inlet Temperature for F-16
- Pre-Test Instructions were to find Highest Vib. Stress Possible and Accumulate 10^7 Cycles
  - Expected to get about 15 ksi-sa in inner panel mode
- Stresses Achieved were 25 ksi-sa with Peaks to 32 ksi-sa for 4.7 million Cycles (Well Above Maximum Engine Running Stresses)
  - Accumulated 5.6 Million Total Cycles Above 13 ksi-sa
  - Two Shrouds Failed Shutting Down Rig
  - 27 of 32 Blades Had Shroud Cracks
- **No Cracks In LSP Treated Area!**

GOODMAN DIAGRAM WITH SIMULATED FOD DAMAGE



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ENDURANCE ENGINE TESTS

- FATIGUE TESTS
- ACCELERATED MISSION TESTS

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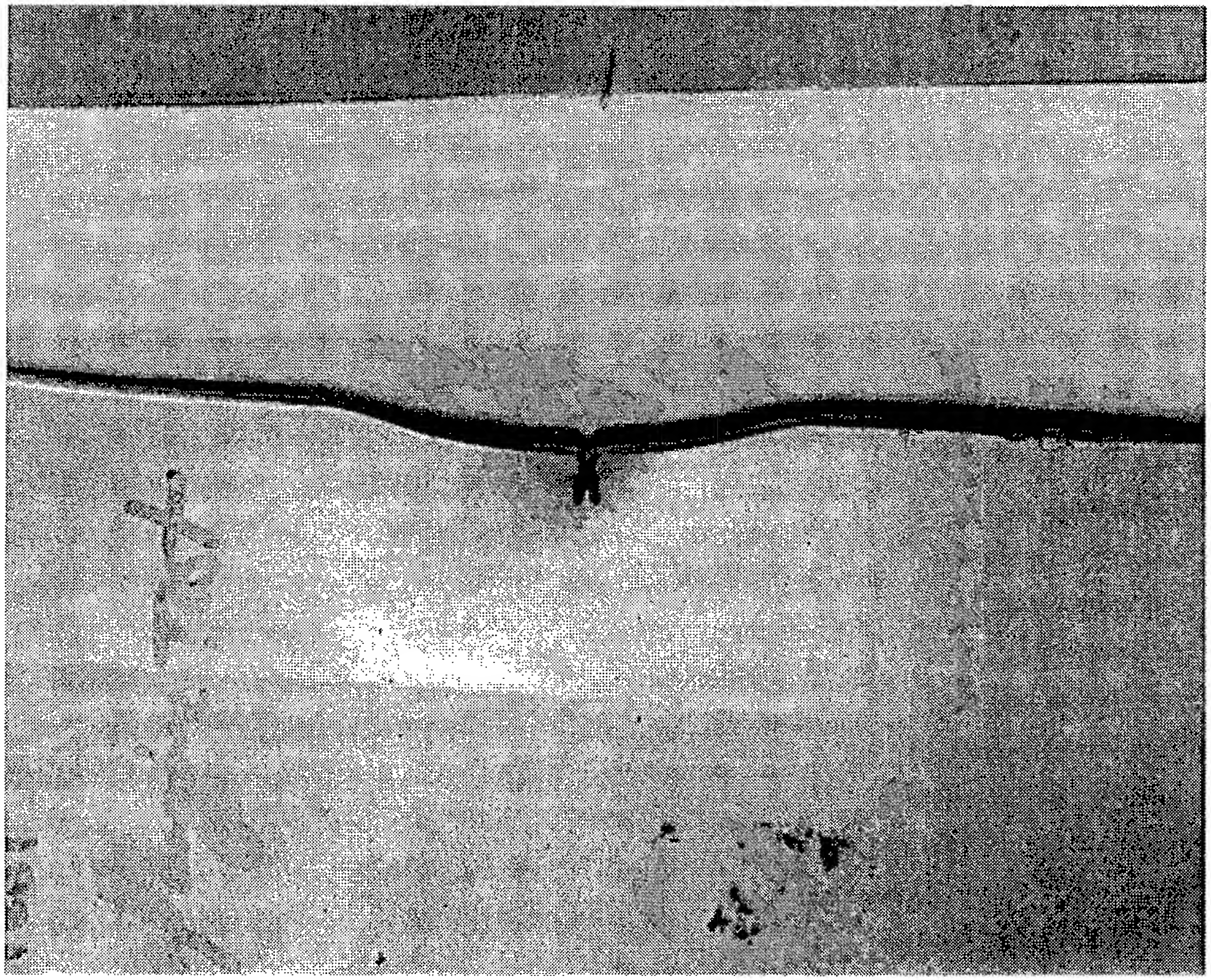
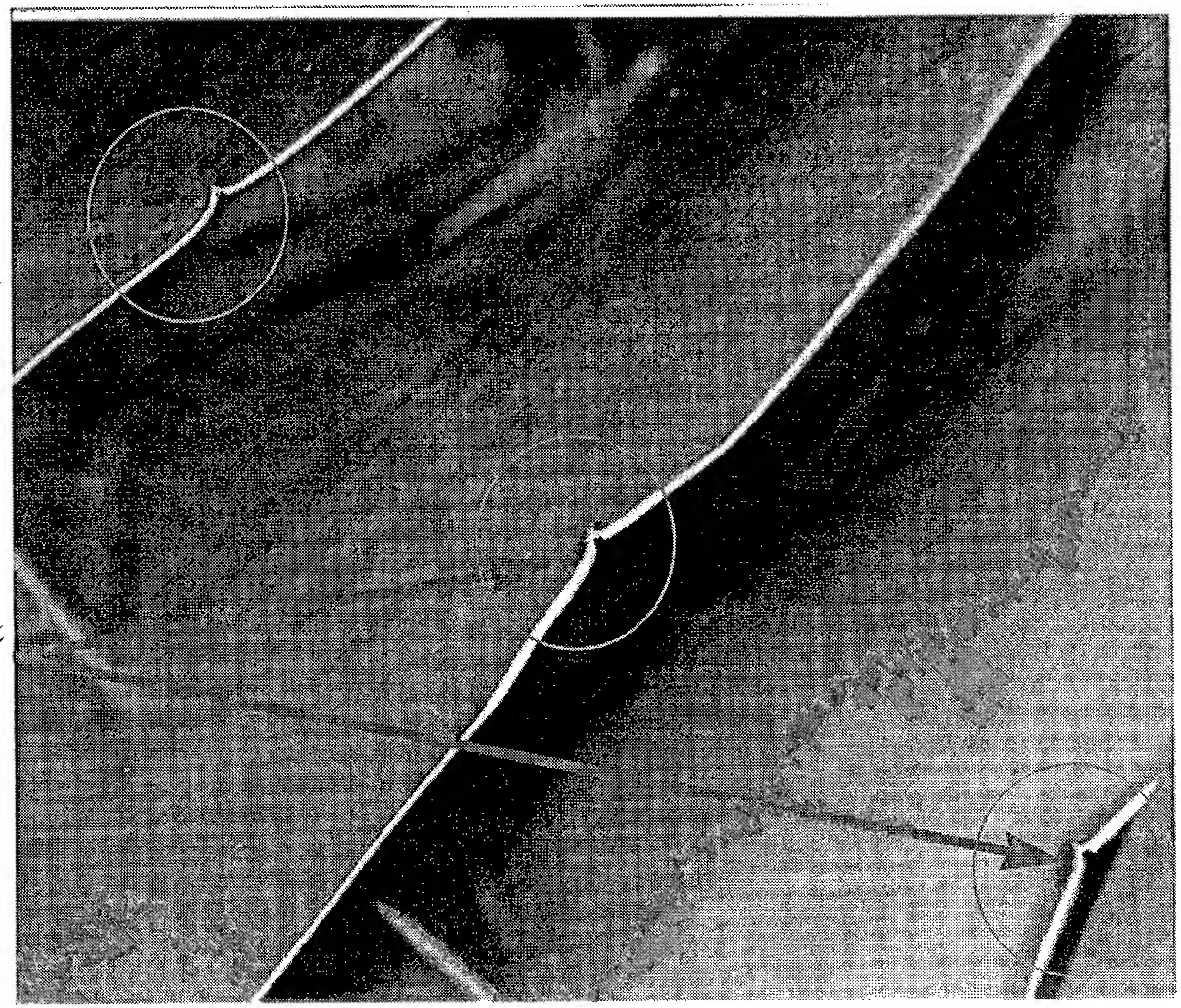
BLENDED & NICKED DAMAGE



ASIP 1997 LSP **W. Cowie, S.R. Mannava, T. Compton: GEAE**

DAMAGED BLADES INSTALLED IN ENGINE



ASIP 1997 LSP

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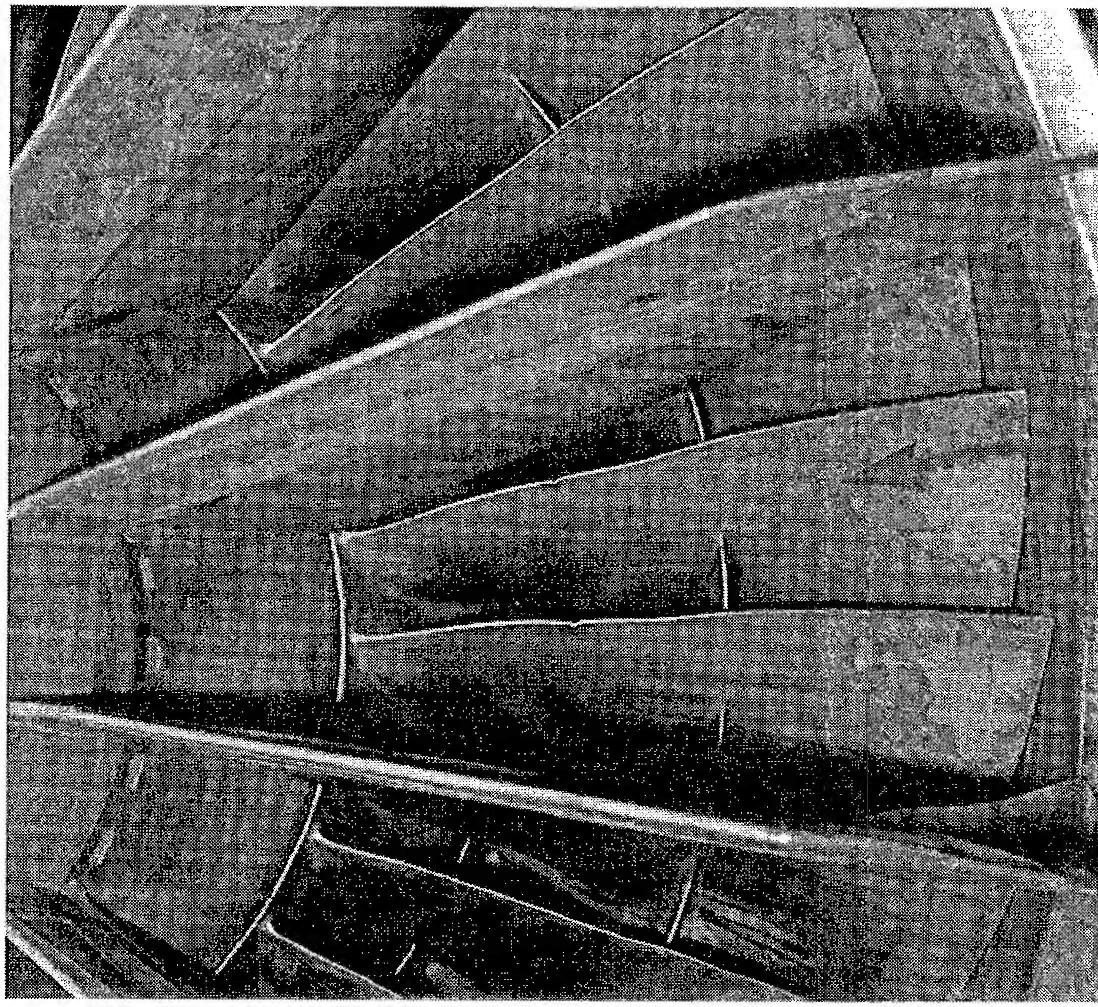
DAMAGED BLADES INSTALLED IN ENGINE



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Upfront HCF Engine Test

- Six Damaged Fan Blades installed in Engine
  - -.060 in. deep V notch at critical location
  - -Two of Six blades were Blended .150 in. and Renicked
- Engine Was Run and Held at Critical Fan Blade Resonance Speed Until Ten Million Cycles Accumulated
  - -Blades not instrumented-Max vibratory stresses driven by Heated Inlet and locked open IGV's
  - -Both conditions increase leading edges' vib stress in damaged area above normal levels
- Blades Inspected at end of Resonance Test- **No Cracks!**

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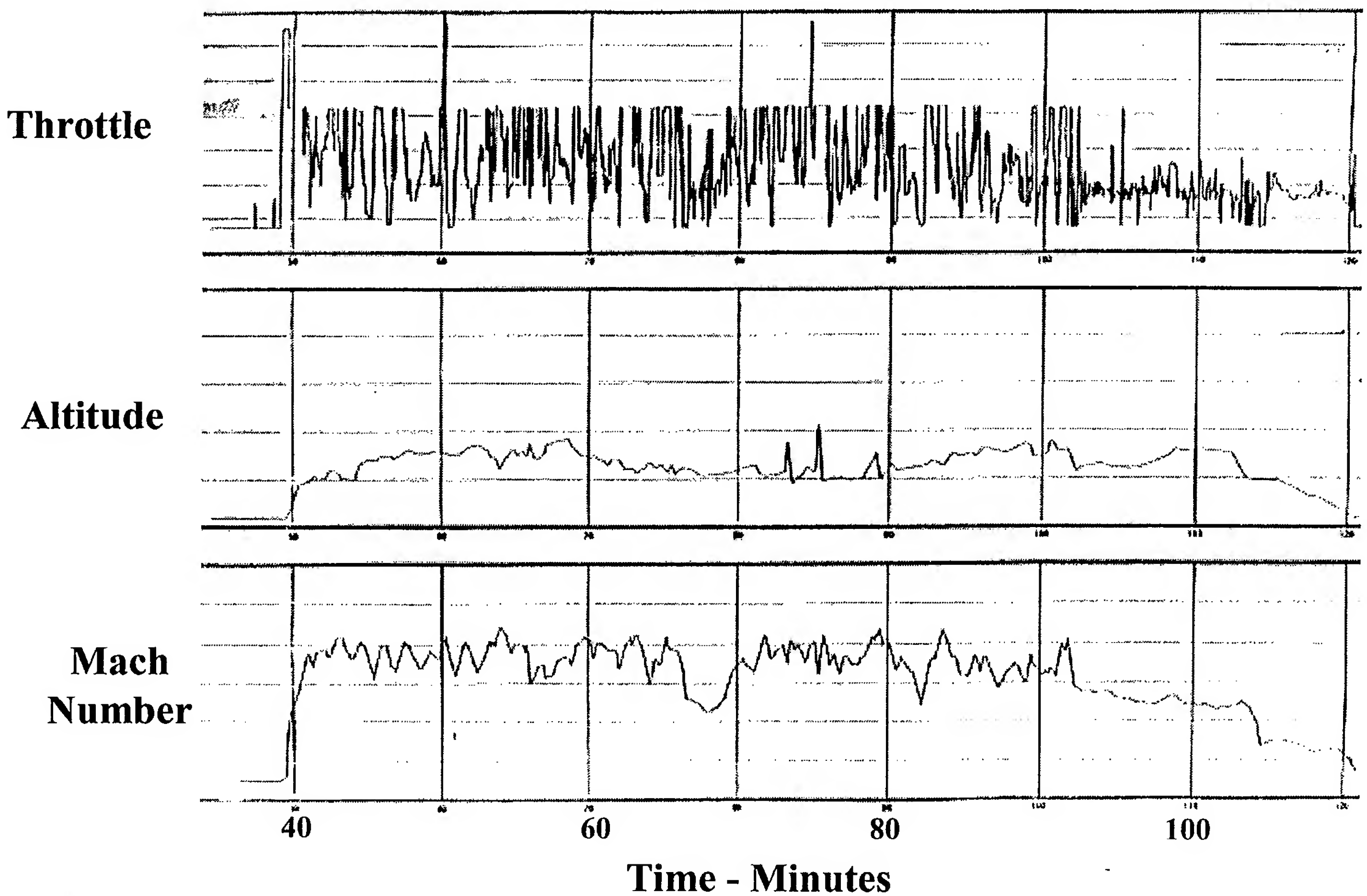
TYPICAL ASIP RECORDER FLIGHT DATA -F16



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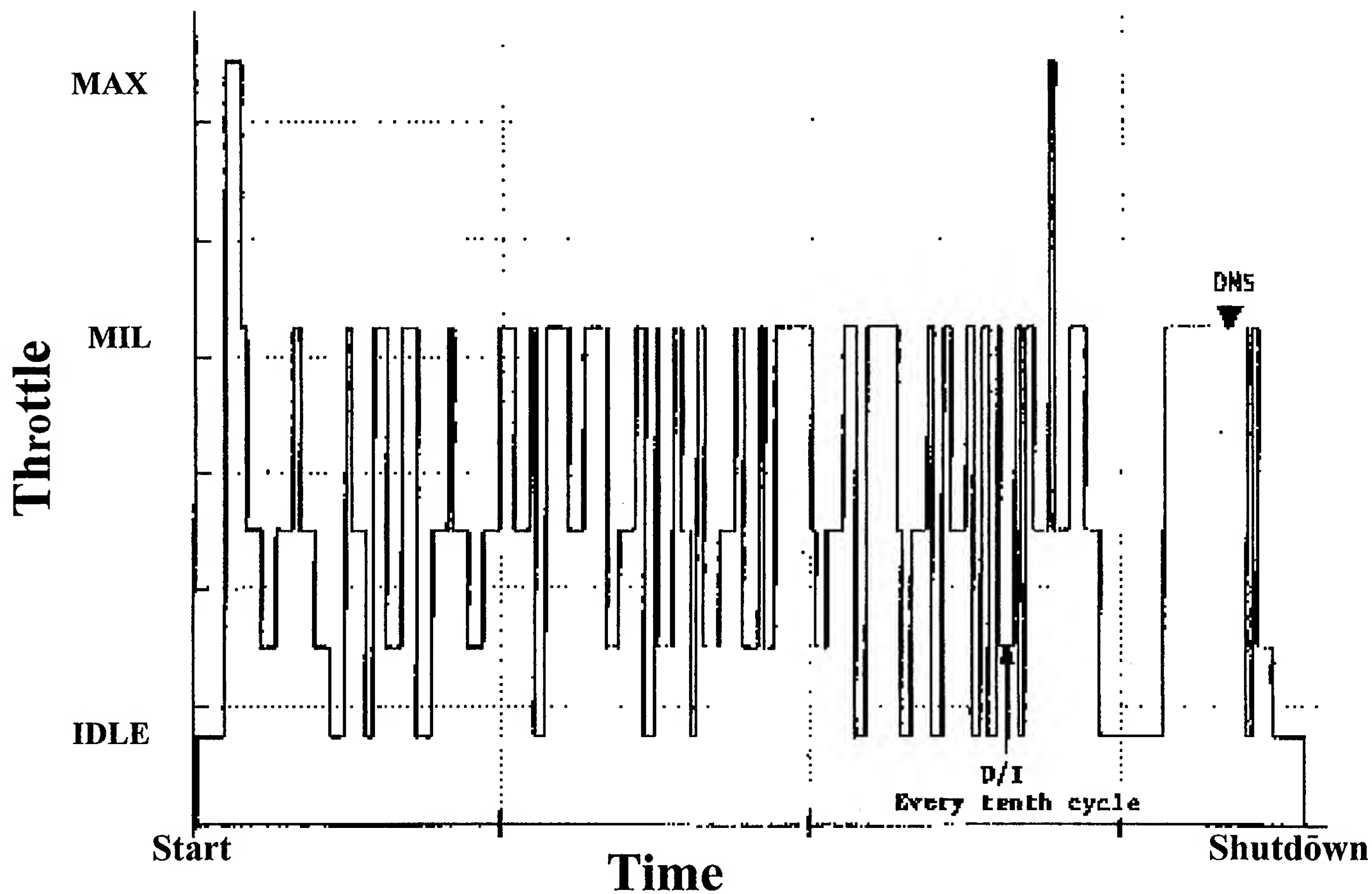
TYPICAL ENGINE ENDURANCE TEST-AGG PROFILE



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Engine Endurance Test

(AMT)

- Simulated 1005 flights with Shutdowns Using Appropriate F16 mission mix and Flight profiles (4000+ TAC cycles)

  -Simulated 805 flights in Ground Test Facility and 200 Flights in Altitude Facility
- Total of 24,880 Throttle Transients- 5-6 years of Field Use

  -Includes 5,370 Throttle Transients at near Mach 1Ram Flight Conditions, 500 ft. Altitude
- Test Time-4 to 6 Months Around The Clock on Automated Throttle- Not counting Inspection Periods
- Damaged LSP Fan Blades remained on the Engine Throughout - Inspection at End of Test Revealed **<u>No Cracks!</u>**

Conclusions

- Damaged LSP Fan Blade Passed Comprehensive Endurance Testing without Cracking
- No Cracking Indicates Even Larger Damage Margin Exists before Blade Failure
- LSP'd F110-GE-129 Fan Blades are Qualified for 1998 Introduction

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LUNCH PRESENTATION

C. Tiffany
Consultant

Aging of US Air Force Aircraft

Charles F. Tiffany
December 11, 1997

NMAB

Outline

- Study Objectives
- Scope of Study
- Committee Membership and Air Force Technical Advisors
- Current Structural Status of the Force
- The Air Force Aging Aircraft Program Objectives
  - Historical Background
- Key Conclusions
  - Aircraft Structural Safety
  - Economics and Readiness
  - Force Management and Service Life Prediction
- Recommended Overall Strategy
  - Description of Strategy
  - Basic Elements of Recommended R&D Program
- Recommended Engineering and Management Tasks
- Recommended Research Opportunities
  - The Prioritization Approach
  - The Prioritized Opportunities

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Scope of Study

Included:

- The structural airframes of 35 types of USAF fixed wing aircraft
  - Air Force supported aircraft
  - Contractor logistics support (CLS), commercial derivative aircraft

Excluded:

- 3 newer aircraft (C-17, F-117, and B-2) - not yet aging A/C
- USAF rotorcraft (H-1, H-53, and H-60)
- Aircraft Engines

Committee

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- Charles Saff, Boeing (former McDonnell Douglas)
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Air Force Supported Aircraft

Aircraft Operator	Aircraft Type	Current Age Data: Years Since IOC	Current Age Data: Average Age (years)	Future Plans
Air Mobility Command	Airlifter and Tanker Aircraft			
	KC-135	41	35	Retain 25+ years. No replacement identified
	C-5	28	18	Retire C-5A in 10–15 years. No replacement identified
	C-141B	32	29	Retire over next 8 years. Replace with C-17
Air Combat Command	Bomber and Attack Aircraft			
	A-10	20	15	Retain 25+ years. No replacement identified
	B-52H	36	34	Retain 25+ years. No replacement identified
	B-1B	11	9	Retain 25+ years. No replacement identified
	F-15	23	12	Retire in 5–20 years. Replace with F-22
	F-16	18	8	Retire in 10–25 years. Replace with Joint Strike Fighter
	Other Aircraft			
	C-130E/H	36	20	Replace 1/3 over 5–25 years with C-130J
	E-3 (AWACS)	20	16	Retire in 17–25 years. No replacement identified
	E-8 (JSTARS)	0	15–20	Retire in 15–20 years. No replacement identified
	EC/AC-135	40	30–35	Retain 25+ years. No replacement identified
	U-2[a]	40	14	Retire in 15–25 years. No replacement identified
	EC-130E/H	36	20	Retire in 15–25 years. No replacement identified
	EF-111	30	29	Retire within next 4–5 years
Air Education and Training Command	Trainer Aircraft			
	T-37B	38	33	Retire in 2–12 years. Replace with JPATS
	T-38	36	29	Retain 25+ years. No replacement identified

[a]This aircraft was developed for the government and is maintained by the manufacturer rather than by an air logistics center.

Force Status
CLS Aircraft

Air Force Designation	Commercial Designation	Quantity	Average Age (years)	Operator(s)
E-4	Boeing 747-200	4	23	ACC
VC-25	Boeing 747-200	2	7	AMC
T-43	Boeing 737-200	13	24	ACC and ANG
C-137	Boeing 707-100/300	6	21	AMC
C-18	Boeing 707-323	6	N/A	AFMC, ACC, USAFA
C-22	Boeing 727-100	3	32	ANG
KC-10	McDonnell Douglas DC-10-30F	59	13	AMC
C-9	McDonnell Douglas DC-9-30	23	26	AMC, USAFE, PACAF
C-12	Beechcraft Super King Air 200	37	17	AFMC, PACAF, AETC
T-1A	Beechjet 400A	156	3	AETC
C-21	Learjet 35A	76	13	All commands
C-23	Shorts 330	3	13	AFMC
C-26	Fairchild SA227 Metroliner	40	5	ANG
C-27	Alenia G-222 Model 710A [a]	10	5	ACC
C-20	Gulfstream II, III, IV	13	10	AMC and USAFE
UV-18	Dehaviland DHC-6 Twin Otter	2	20	USAFA
E-9	Dehaviland DHC-8	2	N/A	ACC
T-3	Slingby T67M260 Firefly	112	3	AETC and USAFA

[a] Not a commercial aircraft, but a military transport originally built for the Italian Air Force.

Utilization Comparison
Air Force vs. Commercial Equivalents

Air Force CLS Aircraft [a]			Commercial Aircraft [a]						
Aircraft	Flights [b]	Hours		Number of Flights			Flight Hours		
			Aircraft	Average	High	Design Goal	Average	High	Design Goal
E-4	7,500–11,000	8,000–10,000	747	~10,000	~32,000	20,000	~40,000	~95,000	60,000
VC-25	N/A	~2,500	747	~10,000	~32,000	20,000	~40,000	~95,000	60,000
T-43	10,000–15,000	16,000–18,000	737	~20,000	~85,000	75,000	~22,500	~80,000	60,000
C-22	51,000–55,000	57,000–59,000	727	~35,000	~72,000	60,000	~47,000	~78,000	60,000
C-18	13,000–44,000	33,000–62,000	707 [c]	~20,000	~37,000	20,000	~40,000	~90,000	60,000
VC-137	8,000–24,000	7,000–52,000	707 [c]	~20,000	~37,000	20,000	~40,000	~90,000	60,000
KC-10	1,400–2,500	6,300–13,000	DC-10	N/A	~36,000	42,000	N/A	~90,000	60,000
C-9	11,600–51,200	11,000–44,600	DC-9	N/A	~99,000	40,000 [d]	N/A	~79,000	30,000 [d]

[a] Approximate data as of 1995 for the commercial aircraft and 1996 for the Air Force CLS aircraft.
[b] Except for the KC-10 and C-9, the data for the Air Force CLS aircraft reflect number of landings, which may be slightly larger than number of flights.
[c] There are 57 707 aircraft remaining in commercial use in the world. There are none registered in the United States.
[d] Contractor revised values to 102,000 flights and 78,000 hours based on retest and tear-down inspection of high-time commercial aircraft.

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Conclusions
AF Challenges

- identify and correct structural deterioration that could threaten aircraft <u>safety</u>
- prevent or minimize structural deterioration that could
  - become an excessive <u>economic burden</u>, or
  - adversely affect <u>force readiness</u>
- for <u>future force planning</u>, predict when it will no longer be viable to retain the aircraft in the inventory
  - excessive maintenance burden
  - extremely poor aircraft availability

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Conclusions
Aircraft Structural Safety

- With increasing age the primary threats to structural safety arise from:
  - the onset of widespread fatigue damage in fail-safe designs
  - the inexorable increase in the number of fatigue-critical areas in safe crack growth designs
- Primary Technical Needs

  Fail-safe designs:
  - improved methods for predicting the onset of WFD
  - NDE methods to rapidly detect small cracks over large areas

  Safe crack growth designs:
  - identification of the next most probable critical area(s)
  - determination of safety limits and inspection requirements for these areas
  - investigation of potential corrosion/environment effects
  - NDE techniques for safety inspections
    - » multilayered structures

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Conclusions
Economics and Readiness

- With increasing age, excessive maintenance costs and reduced readiness arise from:
  - corrosion and stress corrosion cracking detection, repair, and component replacement
  - inspection and repair of fatigue cracks as the number of fatigue critical areas increase and/or if WFD occurs
- Primary technical needs are for the <u>prevention, control, and mitigation</u> of corrosion and SCC
  - detection and rough quantification of hidden corrosion
  - classification of corrosion severity
  - generalized application corrosion-preventive compounds
  - material and process substitution handbook
  - development of technologies for the removal, surface preparation, and reapplication of surface finishes
  - dehumidified storage of aircraft

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Conclusions
Force Management and Service Life Prediction

<u>Air Force Modernization Planning Process (AFMPP)</u>:

- Contains the elements for effective force structure planning and management at all levels
  - needs improved service life inputs

<u>Service Life Prediction</u>:

- The current lack of a comprehensive and accepted service life estimation model is inhibiting Air Force planners from establishing a realistic time table to phase out current systems and begin planning replacement aircraft.
- An economic-based service life estimation model, which accounts for and balances the many cost elements and operational metrics, is needed
  - authoritative guide for supporting replacement decisions and budget inputs
  - analogous to the "COEA" that is performed early to support weapon system milestone decisions

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Conclusions
Force Management (continued)

ASIP and Post-Production Force Management:

- Although ASIP has done a good job of protecting aircraft structural safety for more than 2 decades, the specific factors that have contributed to this past success are being threatened.
  - –reduced budgets
  - –reduced manpower
  - –acquisition changes
  - –increased reliance on contractor maintenance
  - –grade structure limitations at the ALCs
  - –possible complacency of AF management
- Specific factors for ASIP's past success include:
  - rigid enforcement of ASIP by regulation, standards, and specifications
  - implementation of IATPs, which are vital to safety of most military aircraft
  - experienced technical oversight of ASIP activities (e.g. DADTAs)
  - development of competent engineering groups at the ALCs
  - sufficient funding of DADTAs, IATPs, and other ASIP support tasks
  - adequate R&D funding to address design, analysis, inspection, maintenance, and repair needs

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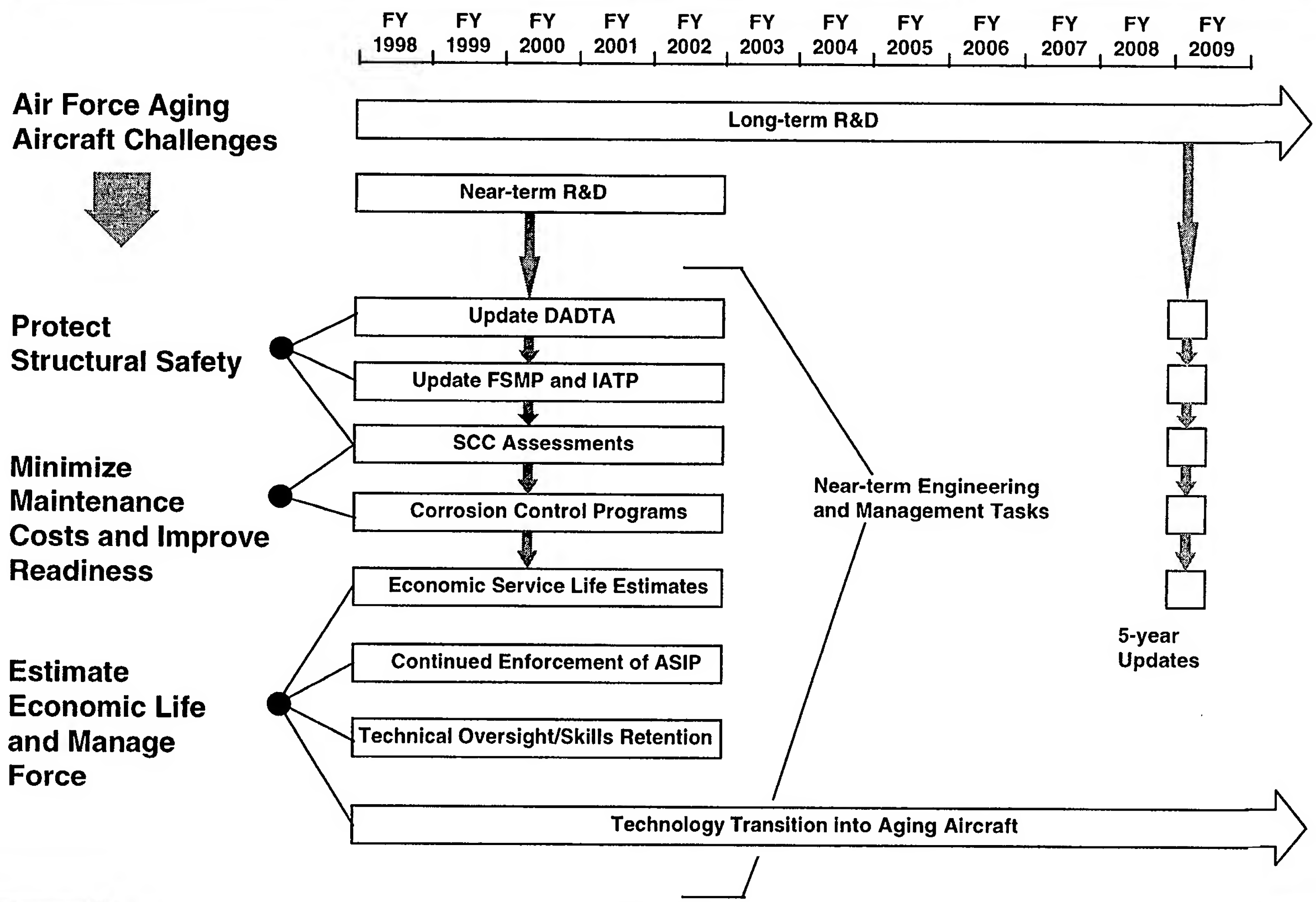
Recommended Overall Strategy



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Basic Elements of Near- and Long-Term R&D Programs

Fatigue
(high- and low-cycle)
–Characterization
–Advanced analysis methods
–Corrosion and environmental effects

Corrosion
–Severity assessment
–Detection and maintenance
–Prevention and control

Stress Corrosion Cracking
–Susceptibility assessment
–Prevention and mitigation
–Component repair and replacement

Challenges
–Protect safety
–Minimize maintenance costs and improve readiness
–Predict economic life and manage aging force

Nondestructive Evaluation
–Detection of small cracks, second-layer cracks, hidden corrosion
–Improve speed and reliability
–Automation

Maintenance and Repair
–Design and analysis methods
–Materials and processes
–Guidelines for component replacement

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Prioritization of DADTA Updates
Air Force Supported Aircraft

Aircraft	Fail-Safe Design	Additional Years in Inventory	Replacement Aircraft Identified	Current Fatigue Cracking	Recent Structural Review[a]	Review Actions Under Way	Priority
KC-135	yes	25+	no	limited	yes	yes	3
C-5A	yes	10–25	no	no report	no	no	2
C-141B	yes	0–8	yes (C-17)	yes	yes	yes	3
A-10	no	25+	no	yes	no	no	1
B-52H	no	25+	no	yes	no	no	3[b]
B-1B	no	25+	no	yes	yes (horizontal tail)	yes	2
F-15	no	5–25	yes (F-22)	limited	no	no	2
F-16	no	10–25	yes (JSF)	yes	yes (fuselage bulkhead)	yes (bulkhead)	1
C-130E/H	yes	25+	some (C-130J)	limited	yes (fuselage)	unknown	2
E-3 (AWACS)	yes	17–25	no	limited	no	no	3
E-8 (JSTARS)	yes	15–20	no	yes	yes (wings)	unknown	2
EC-135	yes	25+	no	limited	yes	yes	3
U-2[c]	no	25+	no	unknown	no	no	1
EF-111	no	<5	no	limited	no	no	none[d]
T-37B	no	0–12	yes (JPATS)	limited	no	no	3[e]
T-38	no	25+	no	yes	no	no	1

[a] Within the past three years.
[b] The lower priority is because a DADTA update was performed in 1995.
[c] This aircraft was developed for the government and is maintained by the manufacturer rather than by an air logistics center.
[d] Based on the assumption that all aircraft will be retired in less than five years as planned.
[e] DADTA is currently being performed by Southwest Research Institute. Update suggested within five years.

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Recommendations
CLS Commercial Derivative Aircraft

Recommendation	Applicable Aircraft	Priority*
(1) For Aircraft that have had DADTAs, update them within the next five years	KC-10, C-27	3
(2) For FAR Part 25 (large transport) aircraft, conduct independent structural reviews to estimate probable (lower bound) onset of WFD.	C-18, C-22, VC-137	1
	C-9	2
	E-4, T-43	3
(3) For FAR Part 23 (utility and commuter) aircraft, conduct a damage tolerance survey to determine need for detailed DADTAs	C-12, UV-18	1
	C-21, C-23, C-20	2
	T-1A, C-26, E-9, T-3	3

* Preliminary based on aircraft age, flight hours, flights, and/or reported potential problem areas

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Recommendations

SCC Assessments and Improved Corrosion Control Programs

<u>SCC Assessments</u>:

- As part of the recommended DADTA updates:
  - identify SCC critical areas
  - evaluate potential failure modes
  - develop alternative corrective actions as necessary

<u>Improved Corrosion Control Programs</u>:

- Each SPD, in concert with the operators and the Air Force Corrosion Control Office should:
  - perform an internal audit of their commercial derivative A/C to ensure full compliance with the CPCPs that have been mandated for their commercial counterparts
  - review the Air Force supported aircraft CPCPs and upgrade them as necessary to a level equivalent to, or better than, those mandated for commercial aircraft
  - evaluate the applicability and cost effectiveness of dehumidified storage of their aircraft

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Recommendations

ASIP, Technical Oversight, and Technology Transition

ASIP:

- The Air Force should facilitate the development of a National Aerospace Standard for ASIP
  - issued by industry and referenced by the government as a measure of acceptable compliance with ASIP requirements.

Technical Oversight:

- HQ AFMC should establish:
  - an aging aircraft engineering resources group to examine and balance available engineering skills and other outside support with aging aircraft needs
  - an aging aircraft technical steering group, chaired by an SAB member, to oversee the recommended engineering and R&D activities
  - 5 technical working groups (1 for each of the elements of the R&D program) to ensure that research meets aging aircraft needs
  - a single knowledgeable and experienced technical leader responsible for the overall aging aircraft engineering and R&D activities

Technology Transition:

- The Air Force should create a seamless funding/budgeting link from development through application and approve only those efforts that have defined transition plans.

Research Recommendations
Priorities

Critical priority: essential to flight safety (i.e., would eliminate a substantial threat to flight safety)

Priority 1: essential to the reduction of maintenance costs and improvement of force readiness (i.e., would enable the Air Force to address significant technical problems)

Priority 2: important to improved flight safety or reduced maintenance costs and improved force readiness (i.e., would represent significant improvements over current solutions)

Priority 3: advantageous to improved flight safety or reduced maintenance costs and improved force readiness (i.e., would improve efficiency and/or reduce cost of current methods)

Research Recommendations

- None With Critical Priority
- Priority-1 (9 total)
  - 4 near term (<5 years)
  - 5 long term (>5 years)
    - » 1 high risk
    - » 2 moderate-high
    - » 2 moderate
- Priority-2 (27 total)
  - 13 near term
  - 14 long term
    - » 4 high risk
    - » 2 moderate-high risk
    - » 4 moderate risk
    - » 4 low risk
- Priority-3 (9 total)
  - 3 near term
  - 6 long term
    - » 2 high risk
    - » 1 moderate-high
    - » 3 moderate
- Monitor Future Structural Issues (Composite Primary Structure)
  - 4 long term

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Research Recommendations
Priority-1

- **Control and Prevention of Corrosion and SCC**
  - near-term research to evaluate and implement improved prevention and control technology
    - » improved alloys and processes
    - » coatings and CPCs
  - long-term basic research to develop materials and processes that significantly improve corrosion and SCC resistance
- **Nondestructive Evaluation Methods**
  - near-term R&D to evaluate, validate, and implement advances in NDE
  - long-term R&D to develop improved quantitative NDE capability and automate wide-area NDE methods

Research Recommendations
Priority-2 and Priority-3

Fatigue

- near-term R&D
  - improve current characterization methods
    - » fail-safe residual strength
    - » onset of WFD
    - » potential effect of environment on safety limits
    - » dynamic response
- long-term R&D
  - develop analytical methods
    - » onset of WFD
    - » high-cycle fatigue behavior
    - » potential effect of environment on short crack growth

Control and Prevention of Corrosion and SCC

- near-term R&D
  - develop corrosion and SCC test protocols
  - methods for early detection of corrosion
  - characterize alloy susceptibility to SCC
- long-term R&D
  - characterize corrosion rates
  - improve SCC prediction methods

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Research Recommendations
Priority-2 and Priority-3 (continued)

Nondestructive Evaluation

- near-term R&D
  - apply advances in automation and data analysis technologies
- long-term R&D
  - hybrid inspection technologies
  - methods and equipment for NDE
    - » corrosion detection
    - » assessment of composite repairs

Maintenance and Repair

- near-term R&D
  - guidelines for fleetwide application of repair advances and materials substitutions
- long-term R&D
  - design and analysis tools for repairs

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Study Summary

- Study completed on time (13.5 months after 1st meeting) and within budget
  - committee conducted 6 meetings and prepared 2 peer-reviewed reports
  - visited Wright Laboratories and San Antonio ALC
- Interim report (completed 3/97)
  - key aging issues and operational needs
  - assessment of Air Force program and plans
- Final report (completed 9/97)
  - summary of force status
  - synopses of aircraft structural histories
  - overall strategy to address structural aging issues
  - recommended
    - » engineering and management actions
    - » prioritized near-term and long-term R&D

December 11, 1997

SESSION III

BONDED COMPOSITE REPAIRS

Chairman - *W. Elliott*
Warner Robins Air Logistics Center

Toward An Expanded Rose Model for Bonded Repair Design and Analysis

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ABSTRACT

The analysis of bonded composite repairs to cracked metal airframes is a complex undertaking due in part to the existence of disbonds. In addition to providing for extremely slow crack growth, the bonded repair must also avoid redistributing structural loads in such a way as to cause a new critical failure to take place. Researchers have identified at least six failure modes that can occur in a bonded repair; all must be addressed in the design and analysis steps. The engineer faced with the choice between a bonded repair and component replacement often chooses the latter option to avoid what appears to be a dauntingly complex analysis task.

The paper describes the ongoing expansion of a closed-form analytical model first developed by L.R.F. Rose to predict the stress state that exists for repaired center cracks in infinite panels. The original model predicted load attraction into the stiffened repair, thermal residual stresses, elastic-plastic shear stresses in the bond line, and the stress intensity factor at the crack tip. The expanded model addresses geometrically non-linear skin bending stresses induced by a single-sided repair, both with and without patch edge tapering and the presence of stiffening elements (substructure), peel stresses in the bond line, and the performance of the repair in the presence of disbonds. These improvements are being incorporated in a highly efficient, accurate, PC-based analytical tool with on-line help and accept/reject criteria. The evolving code allows maintenance engineers to easily and accurately design and analyze bonded composite repairs to complex structures without prior expertise in finite element modeling or bonded repair analysis.

KEY WORDS: Aging aircraft, bonded repair, disbonds, peel stress, Rose model

Overview

- Program Objectives
- Original Rose Model and Additions
- Disbonds: Significant in Bonded Repairs?
- Bending/Peel Contributions to Disbonds
- Conclusions

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Project Objectives

- Enable non-specialists to easily & accurately design, analyze bonded repairs
- Accurately cover all failure modes
  - Expand model to include stiffened structures
  - Explain occurrence of disbonds
  - Understand significance of disbonds
  - Predict crack growth rates adequately
- *Transition technology from fundamental modeling to pragmatic design tools*

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Repairs to load-bearing skin?

Improvements to Rose Model

Original Rose

- Load attraction
- Thermal effects
- Adhesive shear
- Crack bridging
- Constant K
- Infinite, unstiffened sheet

Additional features (in work)

- Tapered patches
- Multiple patches
- Riveted joints
- Effects of stiffeners
- *Secondary bending*
- *Peel stresses*
- *Disbond prediction*

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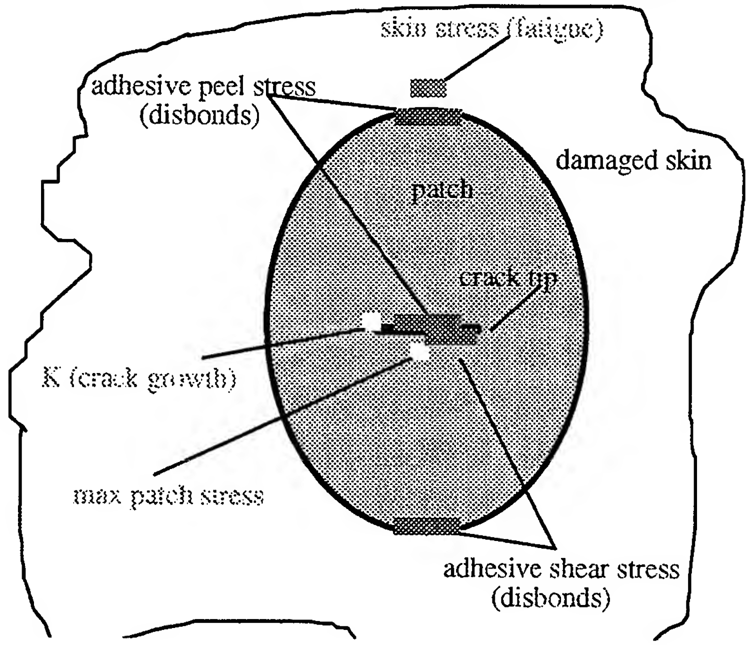
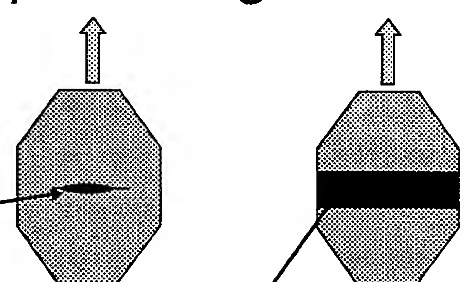
Critical Areas in Bonded Repair



Dis-Bonded Repairs: Significant?



- Small "natural" disbonds observed on crack flanks
- Denney tested large *artificial* disbonds
- Crack growth rates *increase up to 2x* at 20% patch area disbonds, room temperature dry
- Patch effectiveness preserved
- Need accurate da/dN prediction with disbonds

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Bending/Peel Stresses & Disbonds

- Affected components:
  - secondary bending in skin
  - adhesive peel stresses
- Eccentric load paths at joints increase problem complexity
- Stiffened & unstiffened structures affected
- Out-of-plane deformations will drive repair performance

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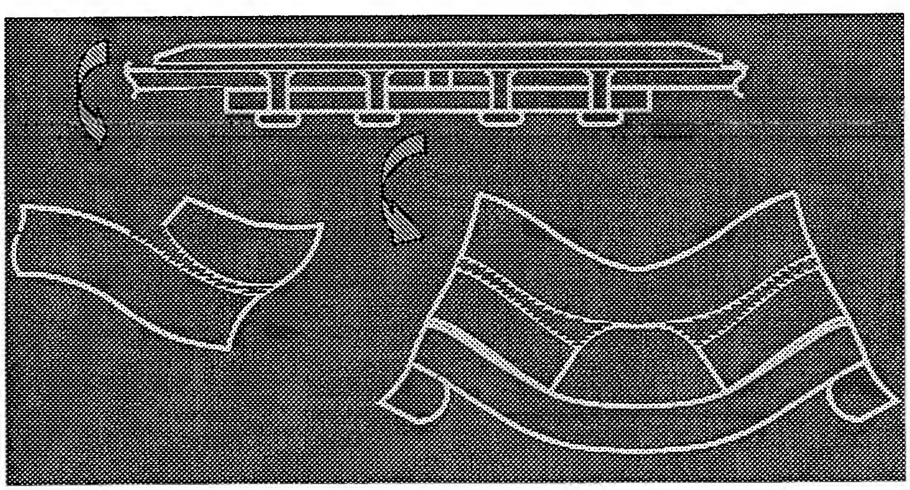
Deformation of butt joint



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Disbond Progression Phenomena

in single-sided, photoelastic materials

unloaded — perfect bond | low load — no disbond | high load — large disbond

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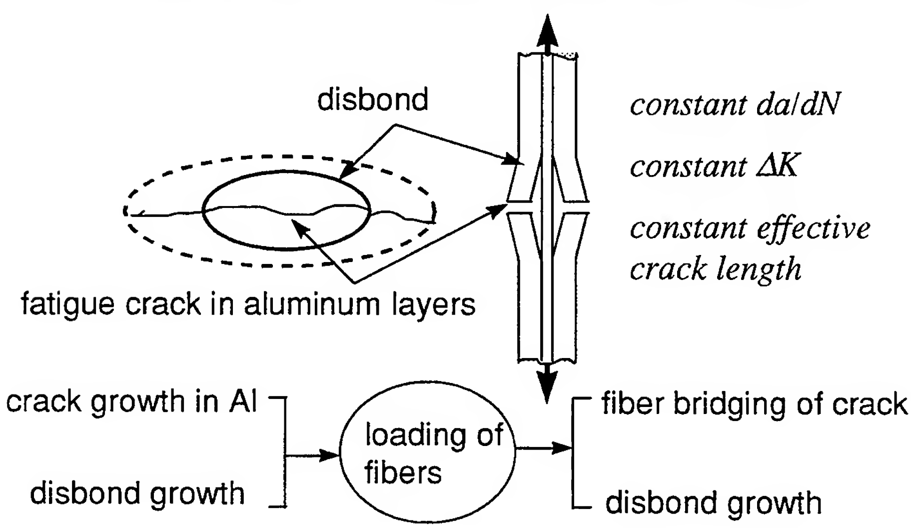
Linked Disbond/Crack Growth Mechanism in Fiber Metal Laminates



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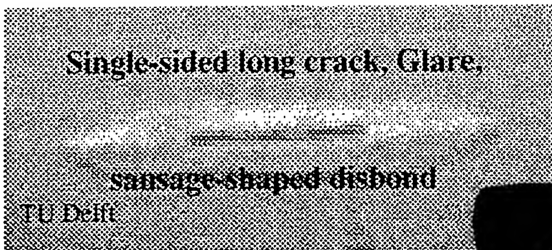
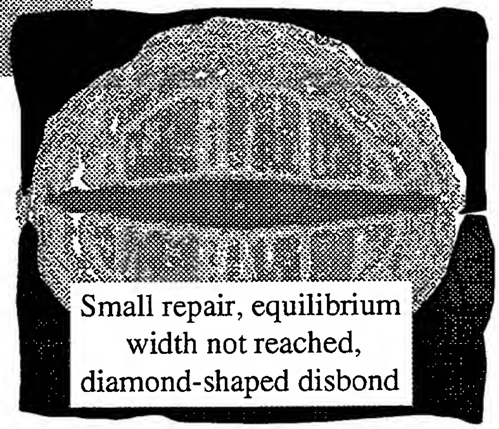
Disbond Morphology



Commonly observed shapes:

- ellipses
- sausages
- diamonds
- .
- .
- .



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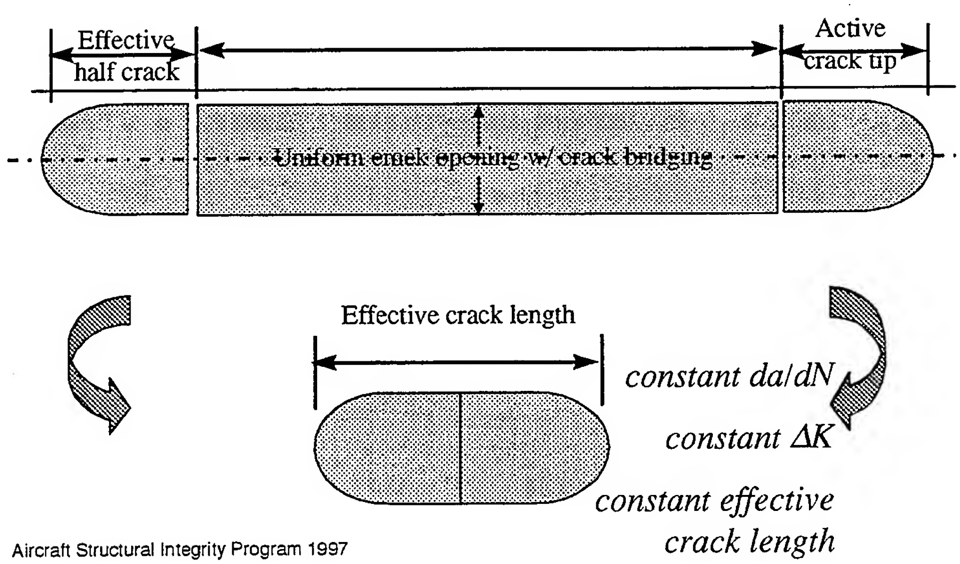
Disbond / Constant K Relation



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Crack Flank Disbonds

- Combination of causes:
  - peel and shear stresses
  - fatigue and environmental degradation
- Peel > shear in single-sided repairs
- Disbond shapes function of
  - combined stress state
  - patch, crack geometries
- Disbond growth arrests when equilibrium reached between bond strength, stress

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Disbond Modeling

- Combined peel and shear drive disbond shape, size in repairs
- Static failure criterion may be useful in predicting disbonds
- Model based on observed sausage-shaped disbonds
- Environmental, fatigue interactions can degrade adhesive properties
  - static knock-down factor?

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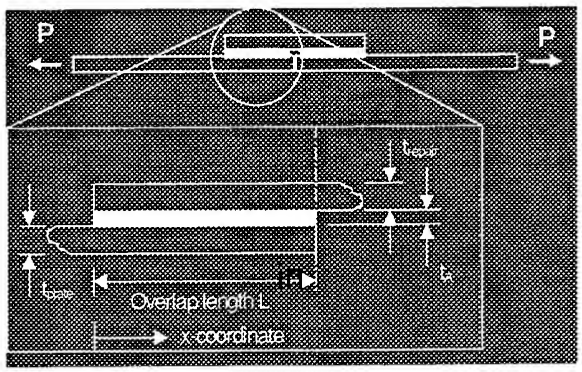
Peel Stress Model

- Closed-form, non-linear solution
- Inputs
  - adherend thicknesses, stiffnesses
  - overlap length
  - adhesive thickness
- Elastic/ perfect plastic
- Predicts shear, peel stresses adhesive



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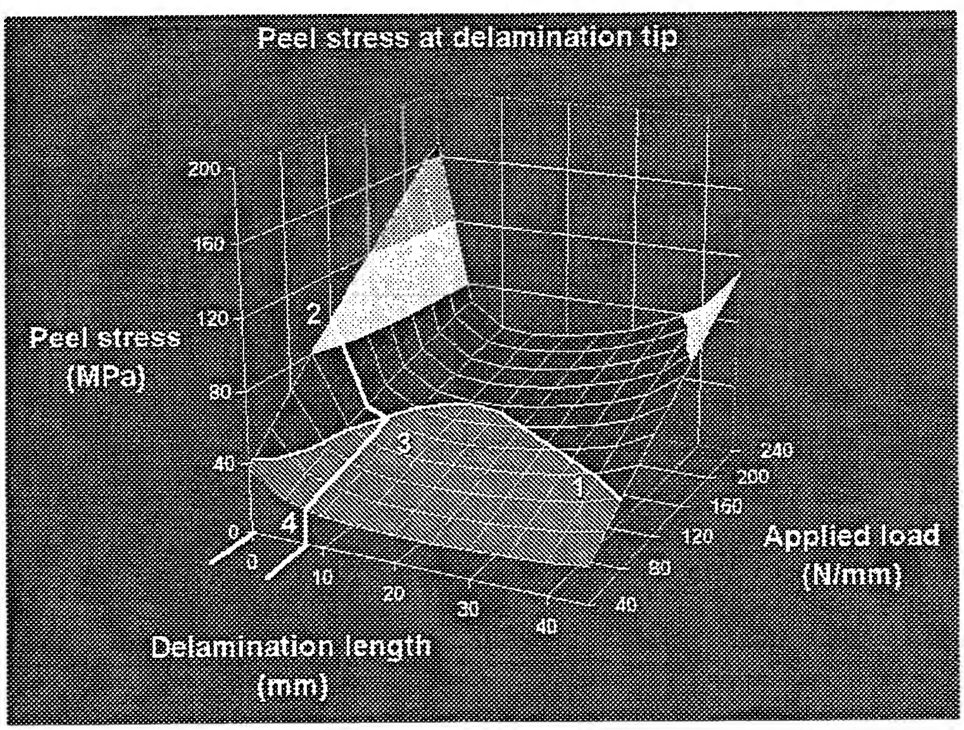

Conclusions

- Peel and shear drive disbond behavior of single-sided repairs
- Growth of disbond depends largely on local peel failure of adhesive
- Disbonds sausage-shaped if crack is sufficiently long
- Single-sided repairs must adequately account for high peel loads
- "Normal" disbond growth acceptable

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Load Eccentricity Key to Disbond Modeling

- Eccentricity is good predictor of:
  - skin bending-induced peel
  - disbond size and growth in single-sided repair
- Combined stresses, fatigue and environmental degradation combine to drive disbond to steady-state
- Relative contributions of each not yet clear
- Good understanding of disbond behavior key to accurate prediction of da/dN

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Environmental Effects on Several Adhesively Bonded Systems

Abstract

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The number of adhesively bonded joints and repairs in aerospace applications is steadily increasing. The advantages offered by bonding over traditional mechanical fastening methods and the growing use of composite materials have served as driving forces behind this increase. Yet, concerns continue to exist regarding the durability of adhesive joints especially with respect to their structural integrity following exposure to and during operation in typical aircraft operating environments. Depending upon the specific aircraft type, these conditions may range from -54°C (-65°F) to 177°C (350°F) with varying amounts of humidity.

In an effort to quantify the performance of some typical adhesively bonded aerospace systems, a fracture mechanics approach was employed. This approach was used because bonded structures may contain flaws in the form of bondline cracks or unbonded regions that serve as initiation points for continued debonding and eventual failure. The systems examined for this project included two that used rubber-toughened epoxies representing bonded repairs and advanced composite structures. A third adhesive, an advanced polyimide being considered for use on future supersonic transports, was also examined.

Bonded joint specimens were isothermally aged under aircraft service conditions for up to 10,000 hours or were exposed to thermal cycling between operational extremes for up to 500 cycles. Following exposure, specimens were tested to determine their Mode I and Mixed Mode I/II fracture toughness and fatigue characteristics. Selected tests were also conducted at temperature levels simulating those encountered during flight operations.

Due to the varying nature of the adhesives examined, environmental effects were pronounced in some cases and nearly negligible in others. However, the research indicated that these effects can be significant and should not be ignored in the design of new bonded structures or the assessment of existing adhesively joined components. An overview of the results of these tests will be presented to illustrate the effects of environmental exposure and operating environments on the fatigue and fracture performance of these bonded systems. These effects will be discussed in terms of the particular application of each specific bonded system and will be used to describe a general approach to incorporating environmental effects into bonded joint design.

Environmental Effects on Several Adhesively Bonded Systems

presented at

The 1997 USAF Aircraft Structural Integrity Program Conference

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2 December 1997

Outline

Goal, Motivation, & Background

Approach

Bonded Material Systems

Adhesives Analyses

- Chemical & Physical Tests
- Tensile & Fracture Tests

Fracture & Fatigue Tests of Bonded Joints

Summary & Conclusions

Goal & Motivation

GOAL

To assess the environmental durability of adhesively bonded joints

MOTIVATION

- Supplement current stress-based design approaches
- Link knowledge of environmental effects with fracture characteristics
- Address USAF, FAA, and industry concerns about specific bonded material systems
- Respond to increased emphasis for life extension of commercial and military "aging aircraft"

Background

Bonded joints have been advocated for aircraft structural fabrication because of a number of advantages:

- weight savings
- improved fatigue resistance
- good sealing capabilities
- compatibility with composites
- better aerodynamics
- long-term cost savings

HOWEVER, adhesive bonding has seen limited use in primary structures due to:

- inadequate surface prep
- difficulty with inspection
- environmentally-induced failures
- costly re-tooling and retraining

Background

A Historical Perspective

- **1910's - 1930's**
  early aircraft - glues and "dopes" for wood & fabric
- **1940's**
  RAF *Mosquito* - plywood construction
- **1950's - 1960's**
  Fokker F-27 and F-28 airliners - bonded aluminum
  USAF B-58 *Hustler* - bonded metal structures
- **1970's**
  USAF *PABST* program - bonded aluminum fuselage
- **1980's - 1990's**
  High performance fighters - bonded composites
  Aging military & commercial aircraft - bonded repairs

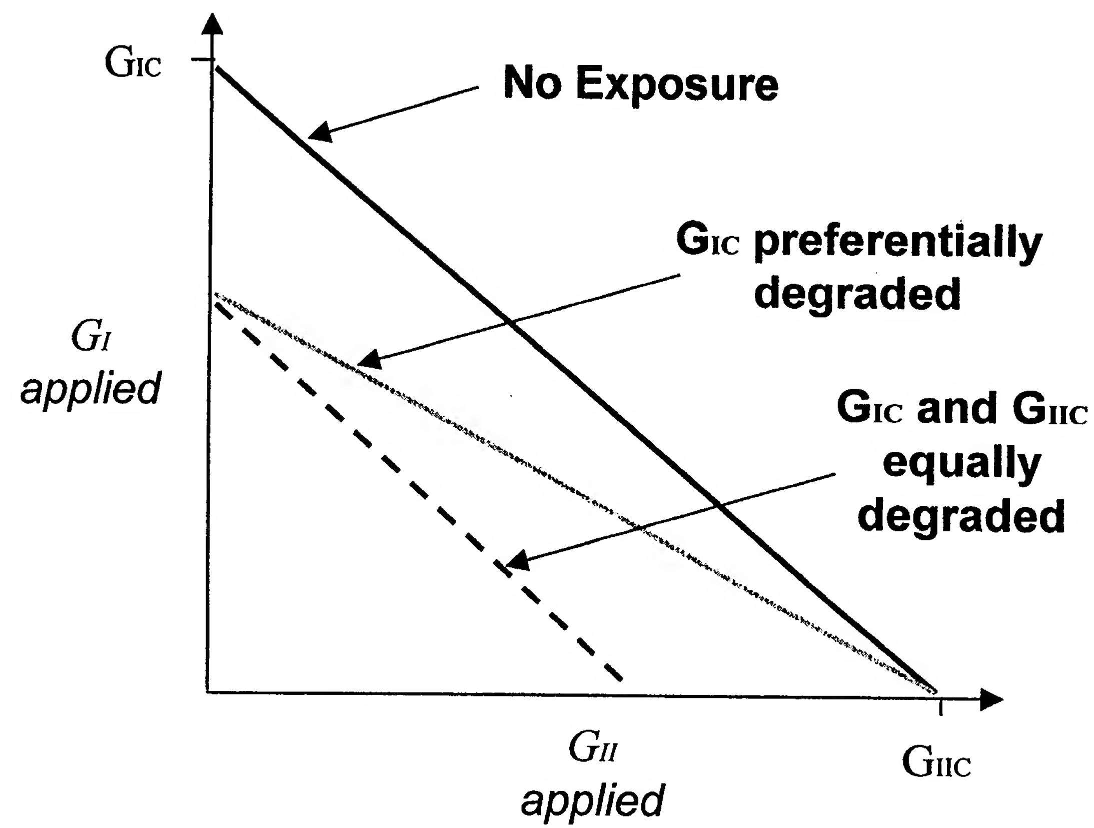
Approach

Provide a broad, general assessment of bonded joint durability using a "building block" approach

- **Investigate existing & future bonded systems in the as-received state and after exposure to service conditions**
  - **Evaluate neat adhesives**
    - chemical, physical, tensile, & fracture properties
  - **Examine bonded joints**
    - Employ fracture mechanics techniques
    - Incorporate finite element analyses if needed
    - Mode I, II and mixed mode fracture properties
    - Mode I fatigue properties
- **Apply experimental and analytical results to applications**

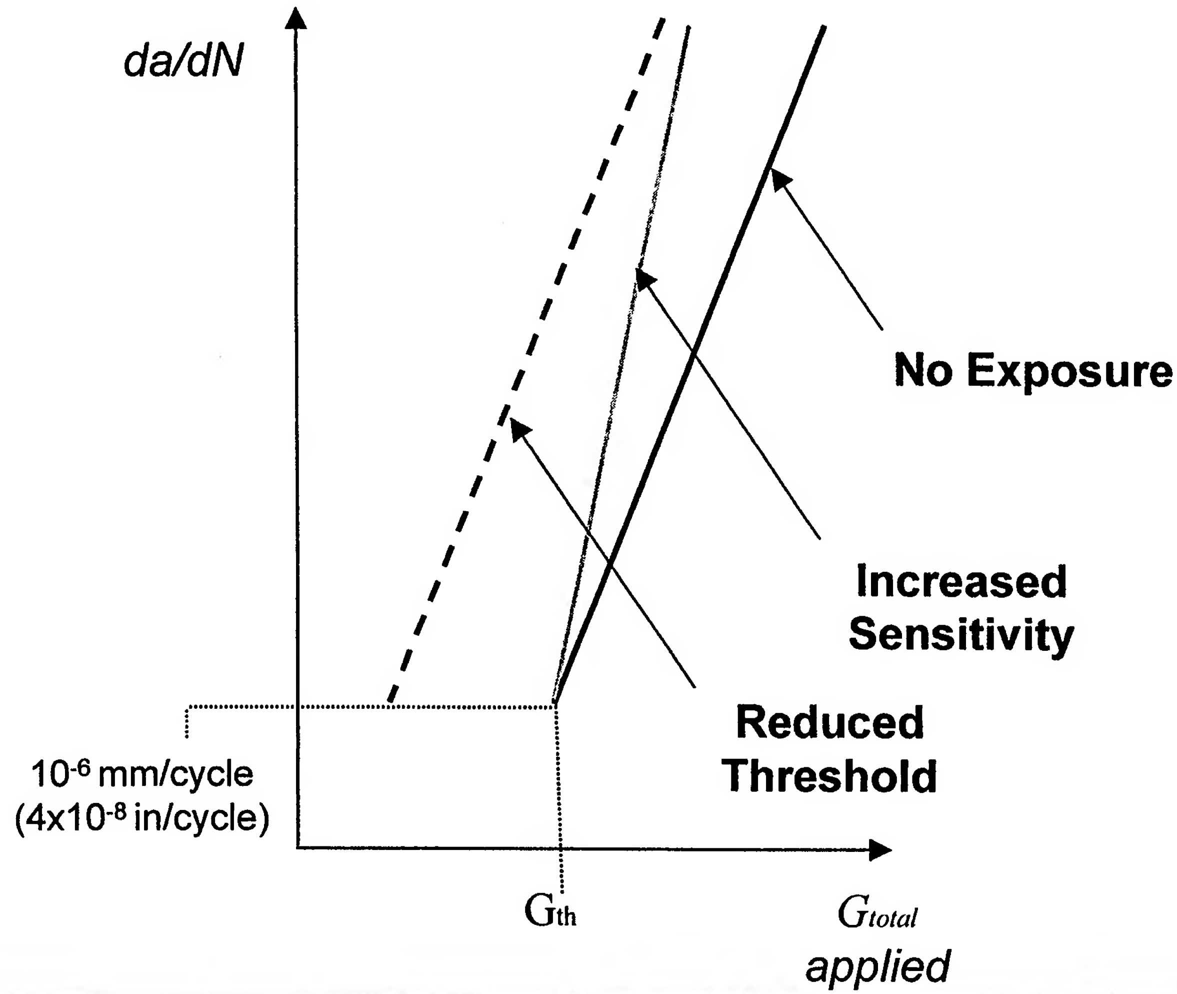
Approach

Possible Effect of Environmental Exposure on Fracture Toughness of Adhesives



Approach

Possible Effects of Environmental Exposure on Fatigue Behavior of Bonded Joints



Materials & Environments

Bonded Repair Systems (Al/FM®73/Al and Al/FM®73/B-Ep)

Manufacturer: *Lockheed Martin*

Application: various aircraft

Adherends:
- 7075-T651 Al/7075-T651 Al [FPL etch]
- 7075-T651 Al/*Textron* F4-5521Boron-Epoxy

Adhesive: *CYTEC* FM®73M toughened epoxy
- non-woven polyester scrim

Environments:
- Minimum Temperature -54°C (-65°F) "cold"
- Maximum Temperature 71°C (160°F) "hot"
- Minimum Humidity 0%rh "dry"
- Maximum Humidity >90%rh "wet"

Materials & Environments

Environmental Conditions Used for the Bonded Repair Material Systems

	100 Cycles	5000 hrs hot/dry	5000 hrs RT/wet	5000 hrs hot/wet	10000 hrs hot/wet	5000 hrs hot/wet + 5000 hrs desiccated	Test @ -54 °C	Test @ 71 °C
FM® 73 U/M Neat Adhesives	X	X	X	X				
Al/FM ® 73M/Al Bonded Joints	X	X	X	X	X	X	X	X
Al/FM ® 73M/B-Ep Bonded Joints	X			X			X	X

Materials & Environments

Bonded Control Surface Systems (Gr-BMI/AF-191/Gr-BMI)

Manufacturer: *Lockheed Martin*

Application: Advanced fighter aircraft

Adherends: *BASF* IM7/5250-4 Carbon-Bismaleimide
- unidirectional and quasi-isotropic

Adhesive: *3M* AF-191M toughened epoxy
- non-woven nylon scrim

Environments:

• Minimum Temperature	-54°C (-65°F)	"cold"
• Maximum Temperature	104°C (220°F)	"hot"
• Minimum Humidity	0%rh	"dry"
• Maximum Humidity	>90%rh	"wet"

Materials & Environments

Environmental Conditions Used for the Bonded Control Surface Systems

	100 Cycles	5000 hrs hot/dry	10000 hrs hot/dry	5000 hrs hot/wet *	Test @ -54 °C	Test @ 104 °C
AF-191 U/M Neat Adhesives	X	X		X		
Gr-BMI/AF-191M/Gr-BMI (uni) Bonded Joints	X	X	X		X	X
Gr-BMI/AF-191M/Gr-BMI (quasi) Bonded Joints	X				X	X

Note: * Hot/Wet exposure conducted at 71°C (160°C), >90% rh

Materials & Environments

Bonded Fuselage System (Ti/FM®x5/Ti)

Manufacturer: *Boeing*

Application: Advanced supersonic transport

Adherends: Titanium (Ti-6Al-4V) [chromic acid anodize]

Adhesive: *CYTEC* FM®x5 amorphous polyimide
- woven glass scrim

Environments:

• Minimum Temperature	-54°C (-65°F)	"cold"
• Maximum Temperature	177°C (350°F)	"hot"
• Minimum Humidity	0%rh	"dry"
• Maximum Humidity	>90%rh	"wet"

Materials & Environments

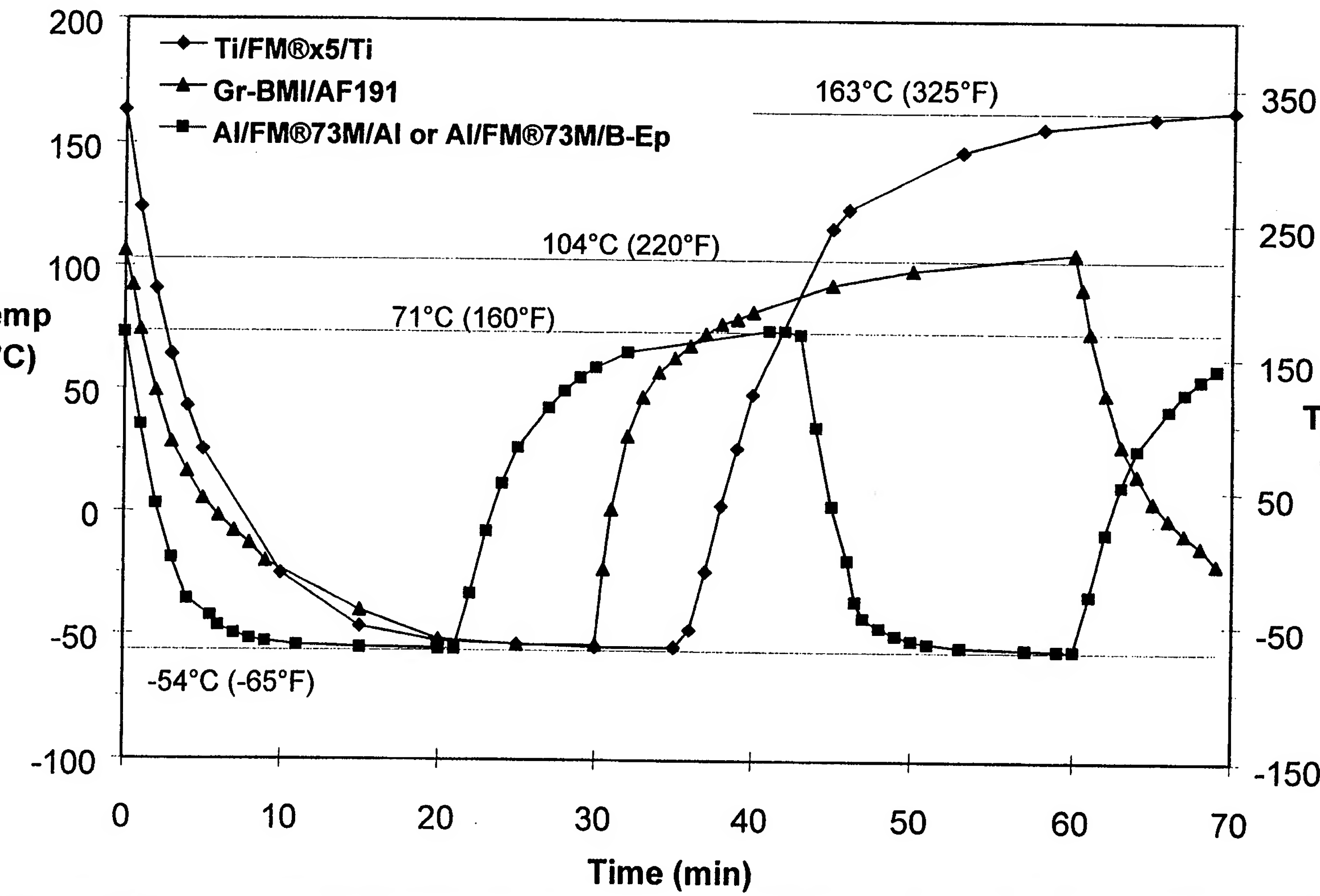
Environmental Conditions Used for the Advanced Bonded Fuselage System

	500 Cycles *	5000 hrs hot/dry	10000 hrs hot/dry	5000 hrs hot/wet **	10000 hrs hot/wet **	Test @ -54 °C	Test @ 177 °C
FM® x5 Neat Adhesives	X	X		X			
Ti/FM® x5/Ti Bonded Joints	X	X	X	X	X	X	X

Notes: * Thermal cycling conducted between - 54°C (-65°C) and 163°C (325°C),
** Hot/Wet exposure conducted at 71°C (160°C), >90% rh

Materials & Environments

Thermal Cycle Profiles



Chemical/Physical Analyses

- **FTIR Spectroscopy**
  - thin adhesive samples illuminated with IR radiation (λ = 1-50 μm)
  - functional groups absorb characteristic frequencies
  - radiation transmitted/reflected gives signature of structure

- **Differential Scanning Calorimetry (DSC)**
  - small amount (~25 mg) of adhesive required
  - specimen heated at same rate as a reference sample
  - DSC device monitors power required to maintain temperature rate
  - heat capacity changes at/near the glass transition temperature, Tg

- **Thermogravimetric Analysis (TGA)**
  - small amount (~10 mg) of adhesive required
  - specimen heated to 900°C (1652°F) while being weighed
  - TGA device monitors weight change as temperature increases
  - significant weight change signals start of degradation or oxidation

Chemical/Physical Analyses

Results

- **Spectroscopy**
  - moisturization detected in epoxies exposed to wet environments
  - possible slight degradation of FM®x5 detected for hot exposures
- **Differential Scanning Calorimetry**
  - ~ 15°C decrease in Tg for:
    - FM®73U and AF-191U exposed to hot/wet conditions for 5000 hrs
    - AF-191U exposed to hot/dry conditions for 5000 hrs
  - no apparent change in Tg of FM®x5
- **Thermogravimetric Analysis**
  - no changes in degradation temperature

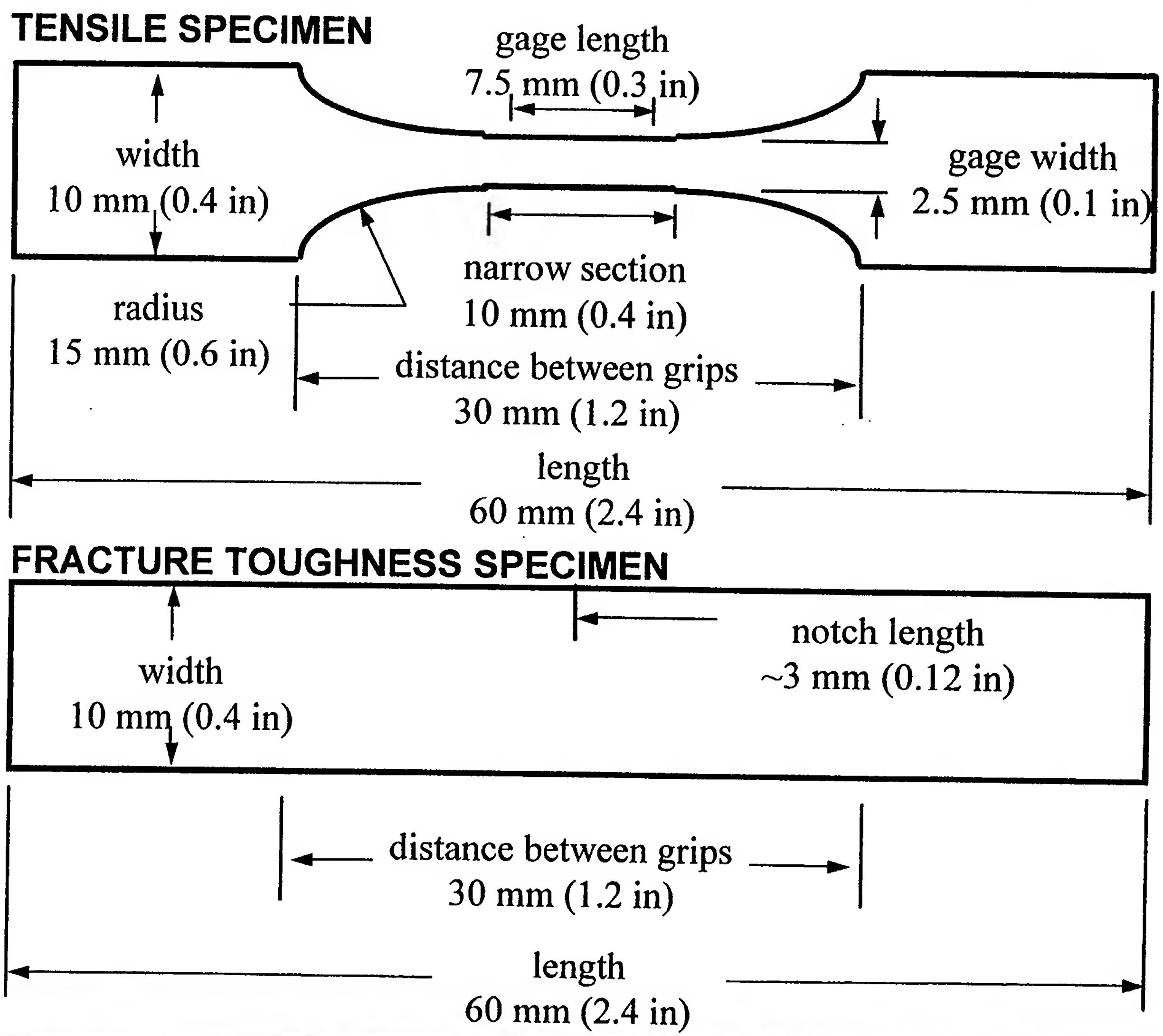
In General: Proved less useful than anticipated

Mechanical Tests of Adhesives

- **Tensile Tests**
  - thin (~0.25 mm [0.10 in.]) neat adhesive film specimens
  - "dogbone" geometry IAW ASTM D638
  - test parameters
    - laboratory conditions, 22°C (72°F), ~50% rh
    - displacement control, 1 mm (0.04 in.)/min.
  - non-contact laser extensometer used for displacement data
- **Fracture Toughness Tests** (plane stress)
  - thin (~0.25 mm [0.10 in.]) neat adhesive film specimens
  - single-edge notched, straight-sided geometry
  - test parameters
    - laboratory conditions, 22°C (72°F), ~50% rh
    - displacement control, 1 mm (0.04 in.)/min.
  - Questar long focal length microscope used to follow crack tip

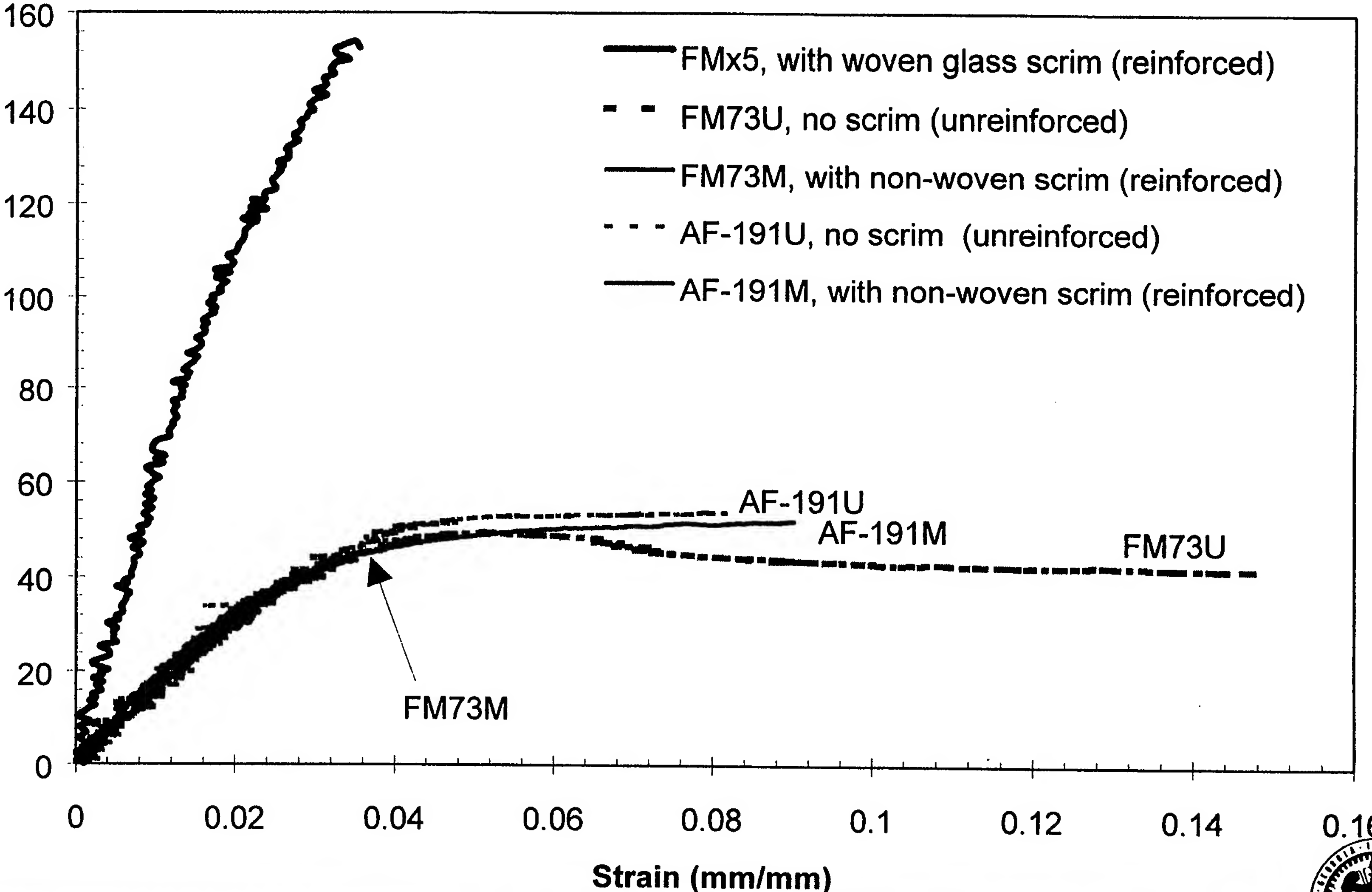
Mechanical Tests of Adhesives

Adhesive Specimen Geometry



Mechanical Tests of Adhesives

Characteristic Tensile Curves for Various Adhesives



Mechanical Tests of Adhesives

Results

- **Tensile Behavior**
  - FM®73U: σ_y & $\sigma_{ult} \downarrow$ ~23% — 5000 hrs RT/wet & hot/wet exposure
    $\varepsilon_f \downarrow$ ~50% — cycling & 5000 hrs hot/dry exposure
  - AF-191U: $\sigma_y \downarrow$ ~16% , $\sigma_{ult} \downarrow$ ~30% — 5000 hrs hot/wet exposure
    $\varepsilon_f \downarrow$ ~56% — 5000 hrs hot/dry exposure
  - FM®x5: σ_y & $\sigma_{ult} \downarrow$ ~25% — 5000 hrs hot/dry exposure
    $\sigma_y \downarrow$ ~40% , $\sigma_{ult} \downarrow$ ~50% — 5000 hrs hot/wet exposure
- **Fracture Toughness**
  - FM®73U: $G_I \downarrow$ ~15% — 5000 hrs hot/dry, RT/wet, hot/wet exposure
  - AF-191U: $G_I \downarrow$ ~15% — 5000 hrs hot/wet exposure
  - FM®x5: no changes observed

Fracture & Fatigue of Bonded Joints

SPECIMENS

- Three specimen geometries for single and mixed mode fracture
- Representative of existing and future materials

ENVIRONMENTAL ISSUES

- "Building Block" approach - Determine G_C and debond growth rates of as-received specimens and: - following environmental exposure (5K and 10K hrs) - following environmental cycling (100 or 500 cycles)

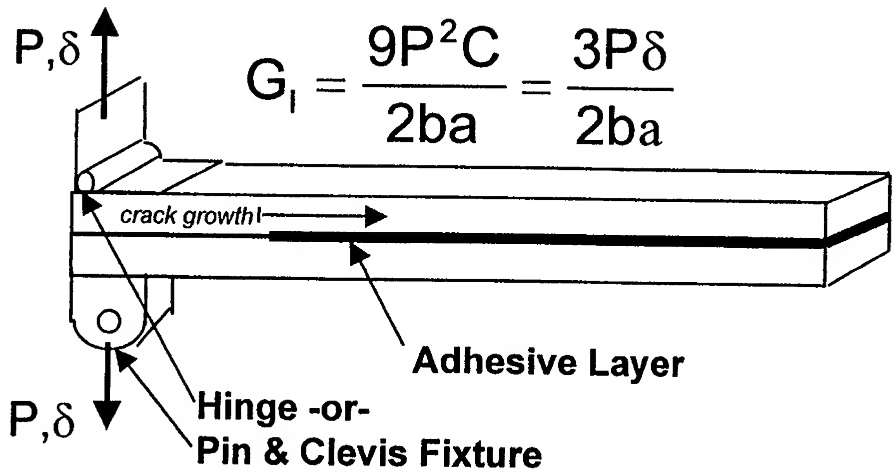
ANALYSIS

- Closed-form and (if needed) finite elements analysis using ABAQUS and GAMNAS

Fracture & Fatigue of Bonded Joints

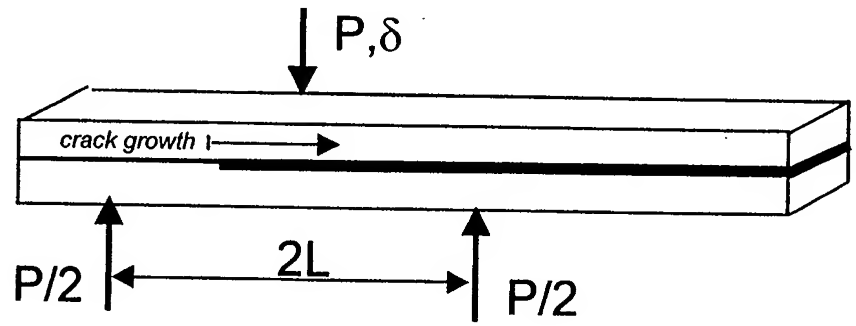
Double Cantilever Beam (DCB)

$$G_I = \frac{9P^2C}{2ba} = \frac{3P\delta}{2ba}$$



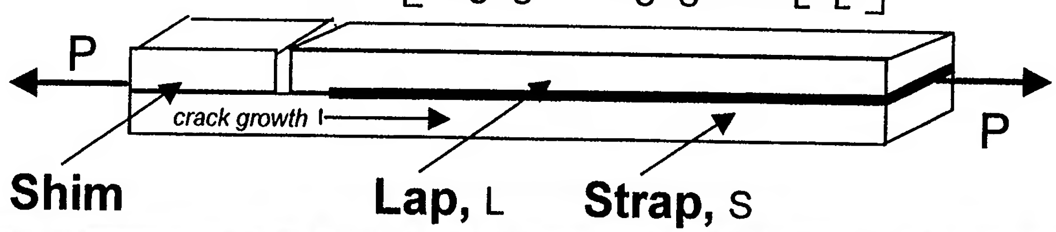
End-Notched Flexure (ENF)

$$G_{II} = \frac{9P^2a^2}{16Eb^2t^3} = \frac{9Pa^2\delta}{2b(2L^3 + 3a^3)}$$



Cracked Lap Shear (CLS)

$$G_T = \frac{P^2}{2b^2}\left[\frac{1}{E_S t_S} - \frac{1}{E_S t_S + E_L t_L}\right]$$

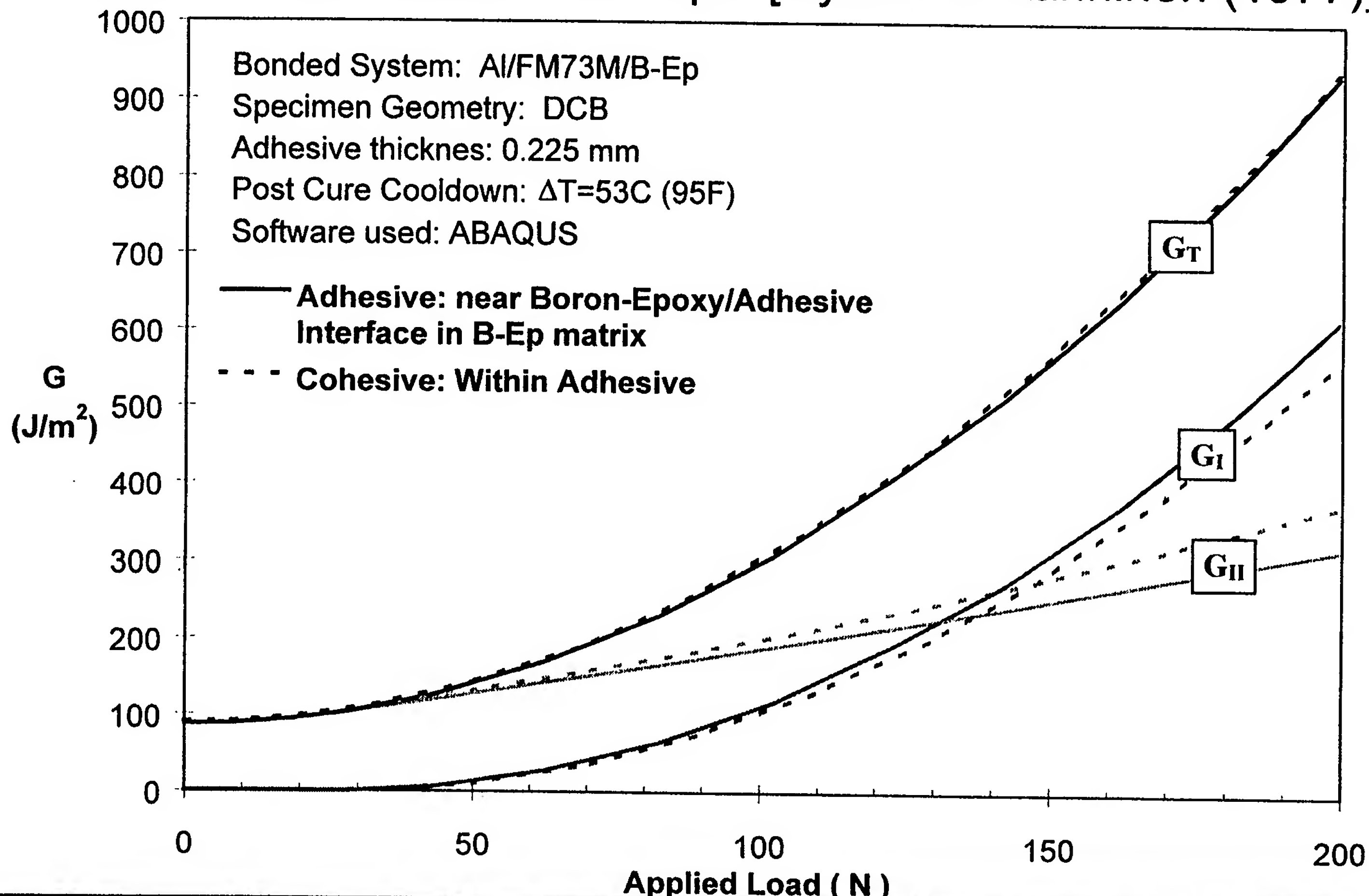


a = crack length
b = specimen width
t = adherend thickness

Fracture & Fatigue of Bonded Joints

Finite Element Analyses (ABAQUS & GAMNAS)

- Required for mixed mode specimens (CLS and all Al/FM®73M/B-Ep)
- Nodal loads and displacements at the crack tip obtained from FEA
- Modified crack closure technique [Rybicki & Kanninen (1977)]



Results from the Al/FM®73/Al System

- **Fracture Toughness**
  - Mode I: 2800 J/m² (16 in·lb/in²) as-received
    - ↓ ~45% tested at -54°C (-65°F)
    - ↓ ~30% 100 thermal cycles, -54°C to 71°C
    - ↓ ~20% 5000 hrs hot/dry exposure
    - ↓ ~70% 5000, 10000 hrs hot/wet exposure
  - Mode II: >1000 J/m² (5.7 in·lb/in²) lower bound
  - Mixed Mode I/II: > 815 J/m² (4.7 in·lb/in²) lower bound

- **Fatigue** (Mode I)
  - Threshold: 100 J/m² (0.6 in·lb/in²) as-received & cycled
    - ↓ ~60% 5000 hrs hot/wet exposure
  - Sensitivity: ~4 ("n" or slope)

Results from the Al/FM®73/B-Ep System

- **Fracture Toughness**
  - Primarily Mode I: 815 J/m^2 (4.7 in·lb/in^2) as-received
    - ↓ ~55% tested at -54°C (-65°F)
    - ↓ ~35% 100 thermal cycles, -54°C to 71°C
    - ↓ ~50% 5000 hrs hot/wet exposure
  - Primarily Mode II: none tested
  - Mixed Mode I/II: >933 J/m^2 (5.3 in·lb/in^2) w/Al strap, lower bound

- **Fatigue** (Primarily Mode I)
  - Threshold: 100 J/m^2 (0.6 in·lb/in^2) as-received & cycled
    - ↓ ~50% 5000 hrs hot/wet exposure
  - Sensitivity: ~10 ("n" or slope)

Results from the Gr-BMI/AF-191 System

- **Fracture Toughness**
  - Mode I: 1720 J/m^2 (9.8 $in \cdot lb/in^2$) [uni], as-received
    1490 J/m^2 (8.52 $in \cdot lb/in^2$) [quasi], as-received
    ↓ 10 %[uni] tested at -54°C (-65°F)
    ↓ ~25 %[uni] 5000, 10000 hrs hot/dry exposure
  - Mode II: ~3000 J/m^2 (17 $in \cdot lb/in^2$) [uni], all conditions
    >2850 J/m^2 (16.2 $in \cdot lb/in^2$) [quasi], lower bound
  - Mixed Mode I/II: ~6000 J/m^2 (34 $in \cdot lb/in^2$) [uni], as-received
    >2500 J/m^2 (14.4 $in \cdot lb/in^2$) [quasi], as-received
    ↓ >40 %[uni & quasi] tested at -54°C (-65°F)
- **Fatigue**
  - Threshold: 100 J/m^2 (0.6 $in \cdot lb/in^2$) all conditions
  - Sensitivity: ~6 ("n" or slope)

Results from the Ti/FM®x5/Ti System

- **Fracture Toughness**
  - Mode I: 2500 J/m² (14.4 in·lb/in²) as-received
    - ↓ ~20% tested at -54°C (-65°F)
    - ↓ ~35% 5000-10000 hrs hot/dry exposure
    - ↓ ~45% 10000 hrs hot/wet exposure
  - Mode II: ~6000 J/m² (34 in·lb/in²) inconsistent crack growth
  - Mixed Mode I/II: ~1100 J/m² (6.3 in·lb/in²) interfacial failure

- **Fatigue** (Mode I)
  - Threshold: 100 J/m² (0.6 in·lb/in²) all conditions
  - Sensitivity: ~4 (“n” or slope)
  - large amount of scatter

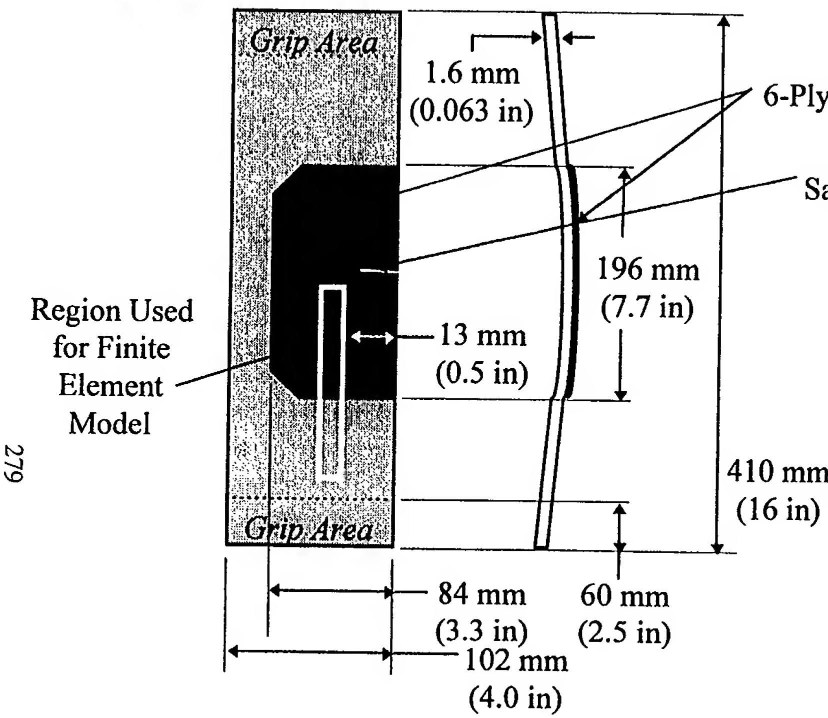
Case Study

Boeing/Textron Bonded Doubler Program

- **Boeing & Textron Efforts**
  - Investigated fatigue of patched panels
  - Used the Al/FM®73M/B-Ep system
  - Found no patch debonding after 300,000 cycles under typical flight loads (21-138 MPa [3 to 20 ksi])
- **Georgia Tech Efforts**
  - Independent study
  - Modeled a portion of the bonded doubler as a CLS specimen
  - Accounted for thermal mismatch, residual curvature, crack location & size, patch taper, and grip type (i.e. boundary conditions)
  - Found ΔG_T for the Boeing/Textron tests ~ 40 J/m^2 (0.23 $in.lb./in.^2$)
  - This low ΔG_T consistent with lack of observed debonding

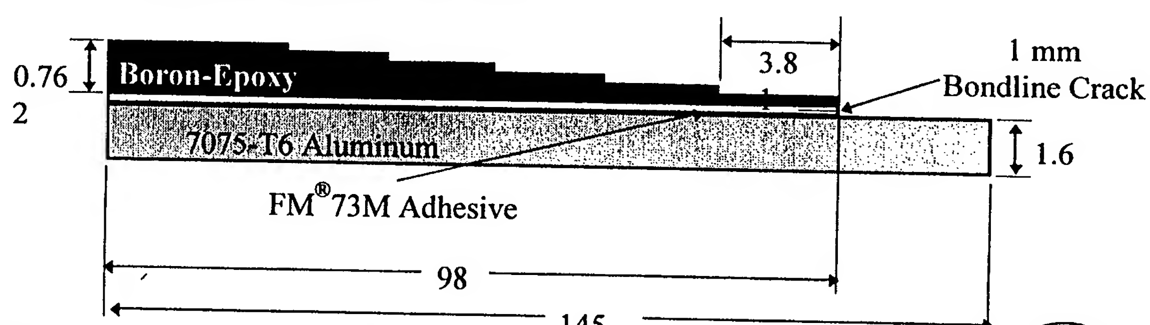
Case Study

Boeing/Textron Bonded Doubler Program



Boeing/Textron Test Specimen

- All measurements in mm
- Boron-epoxy ply thickness: 0.127 mm
- Adhesive thickness: 0.225 mm



Georgia Tech Model

Summary

- **Fracture & Fatigue Tests of Bonded Joints**
  - **Fracture**
    - *General* trend of decreased toughness with long term exposure
    - No universal relationship among Mode I, Mode II, and mixed mode toughness values
  - **Fatigue** (based on results from DCB specimens)
    - Threshold levels ~ 100 J/m^2 (0.57 in.lb./in.^{2})
      - sometimes reduced by long term exposure
    - Growth rate sensitivity ~ 2-6 times greater than metals
      - unaffected by exposure
  - **Crack Paths** (fracture surfaces)
    - Al/Al/FM®73M/Al: cohesive
    - Al/FM®73M/B-Ep: in composite matrix near adhesive/B-Ep interface
    - Gr-BMI/AF-191M: dependent upon adherends & geometry
    - Ti/FM®x5/Ti: “mixed” cohesive (except adhesive in CLS)

Summary

- **Environments**
  - Changes were detrimental
  - Testing at -54°C (-65°F) & hot/wet pre-test exposure most severe
  - Thermal cycling resulted in some degradation
- **Comparing the Bonded Systems**
  - Toughened epoxies (FM®73 and AF-191) behaved similarly
  - Polyimide (FM®x5) was most resistant
  - Metal surface prep provided adequate protection against environmental conditions evaluated
- **Case Study** (Boeing/Textron bonded doubler)
  - Simulated flight loads resulted in ΔG_T below threshold levels for geometry and conditions investigated

Conclusions

Fracture mechanics proved useful in assessing durability

Exposure to service environments can be detrimental

Changes in toughness & fatigue behavior of bonded joints:

- are different for different bonded material systems
- are not always the result of poor surface preparation
- are not always indicated by chemical or physical tests of adhesives
- may be reflected in the mechanical behavior of neat adhesives

Fracture path depends on adhesive, interface, & adherend

A safe life design approach for bonded joints appears best:

- low threshold values with respect to monotonic fracture toughness
- extremely high growth rate sensitivities

Use of experimental data and finite element analysis shows promise in bonded joint design evaluation

Acknowledgments

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Committee Members:	**Prof. David McDowell** School of Mechanical Engineering
	Prof. Richard Neu School of Mechanical Engineering
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	Mr. James Rudd Wright Laboratories, Wright-Patterson AFB, OH
Sponsor:	**Federal Aviation Administration Technical Center**
Technical Monitor:	**Don Oplinger**
Grant Number:	**95G023**
Material Suppliers:	**Lockheed Martin** Marietta, GA **3M Corp.** St. Paul, MN **Boeing** Seattle, WA **CYTEC** Havre de Grace, MD

Qualification of a Bonded Repair to C-5A Fuselage Cracking Under Spectrum Fatigue Loading

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ASIP 1997

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Lt. Jim Ryan

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Prof. J. Schijve
Delft University of Technology

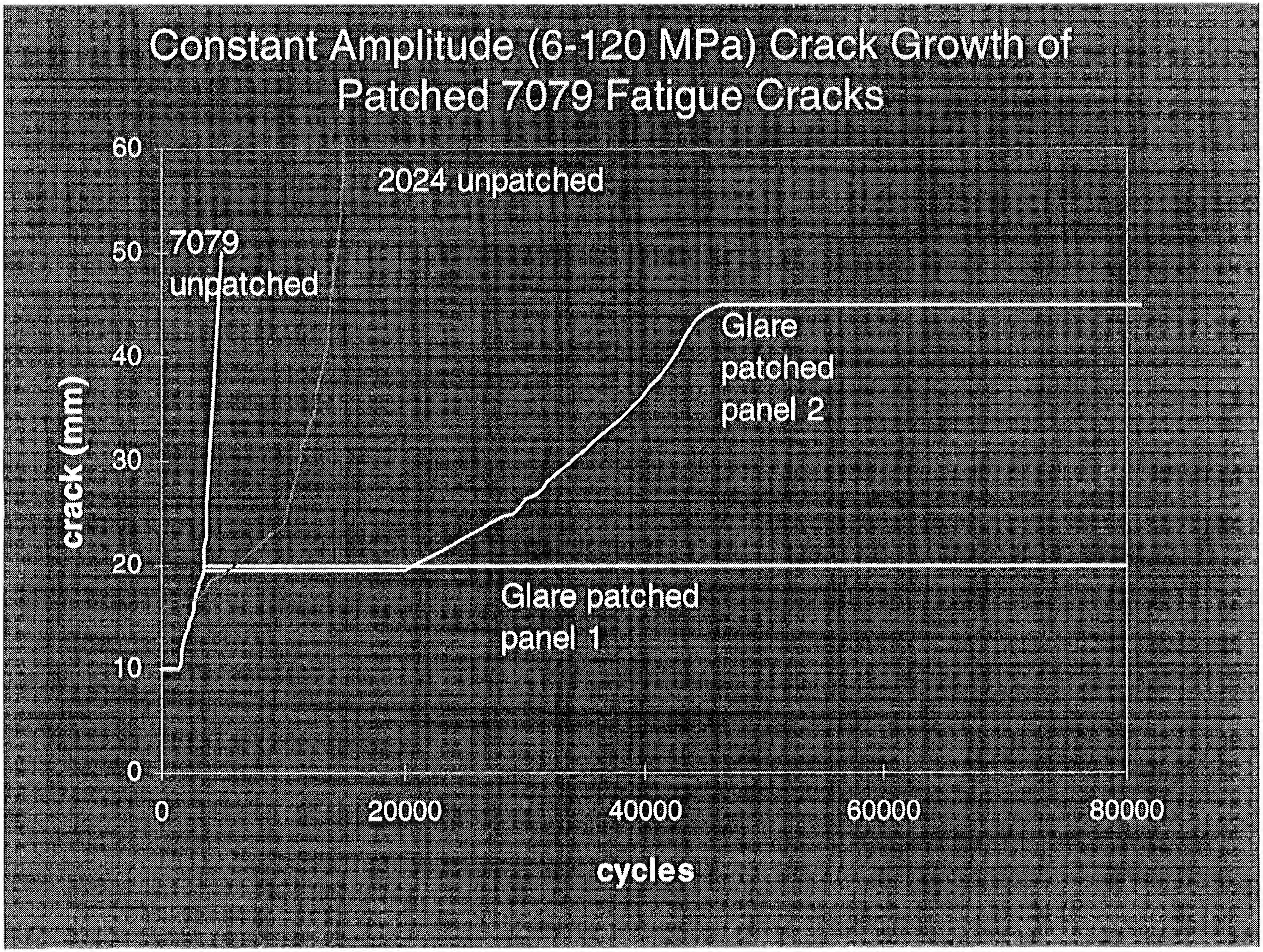
Maj. Rob Fredell, USAFA

Overview

- C-5A Fuselage Crown Cracking
- Spectrum Loading for Bonded Repairs
- Test Spectrum Development
- Test Results
- Conclusions

C-5A Fuselage Crown Cracking

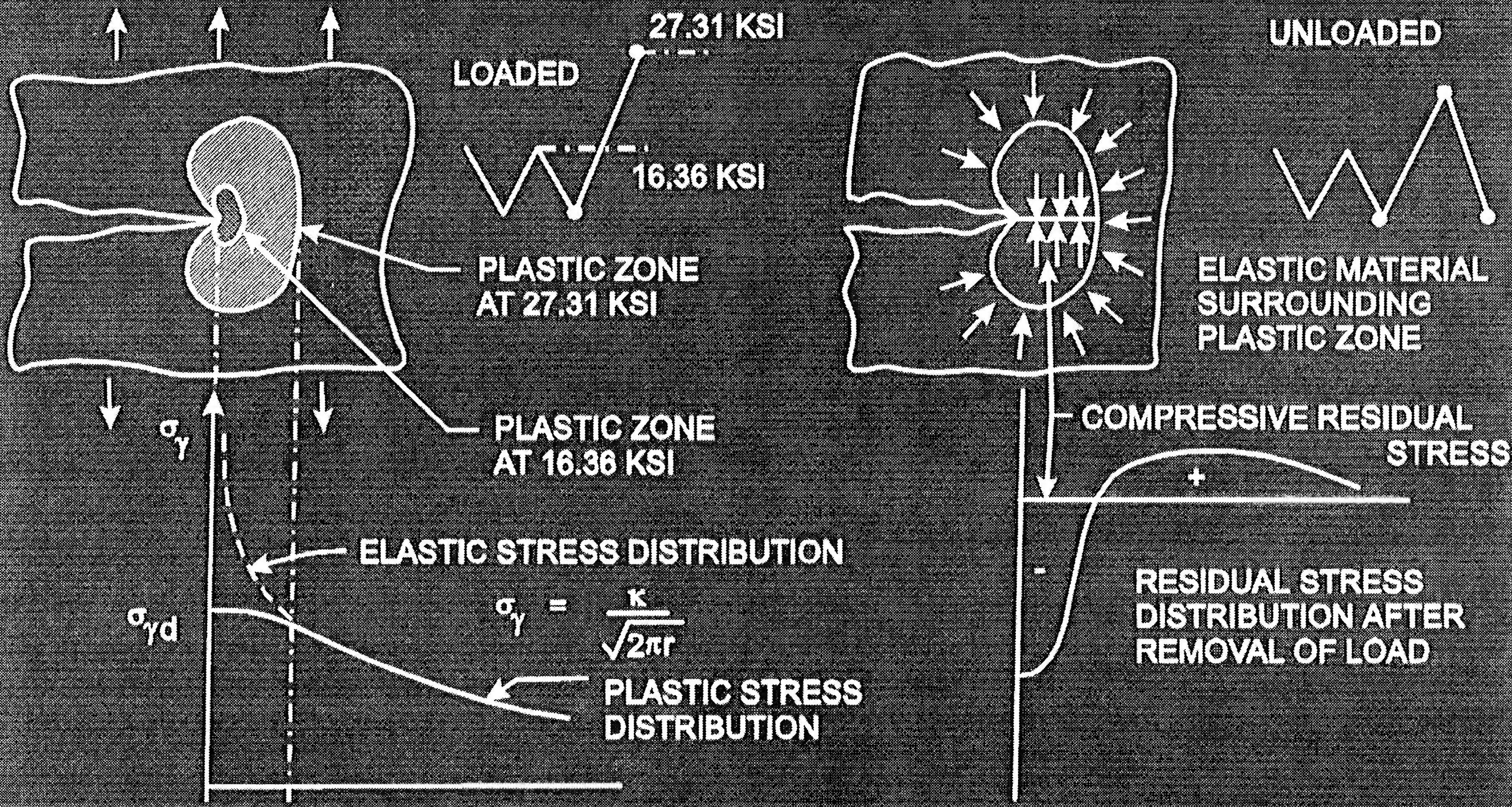
- Thin structures
- Operating temperatures -55 C
- Highly loaded crown (tail bending)
- Poor fatigue properties of 7079-T6
- Mechanical fastened repairs can create new problems
- Reskinning complicated
- Prototype bonded repairs applied in Oct 1995


Constant Amplitude (6-120 MPa) Crack Growth of Patched 7079 Fatigue Cracks
crack (mm)
0
10
20
30
40
50
60
cycles
0
20000
40000
60000
80000
2024 unpatched
7079 unpatched
Glare patched panel 2
Glare patched panel 1

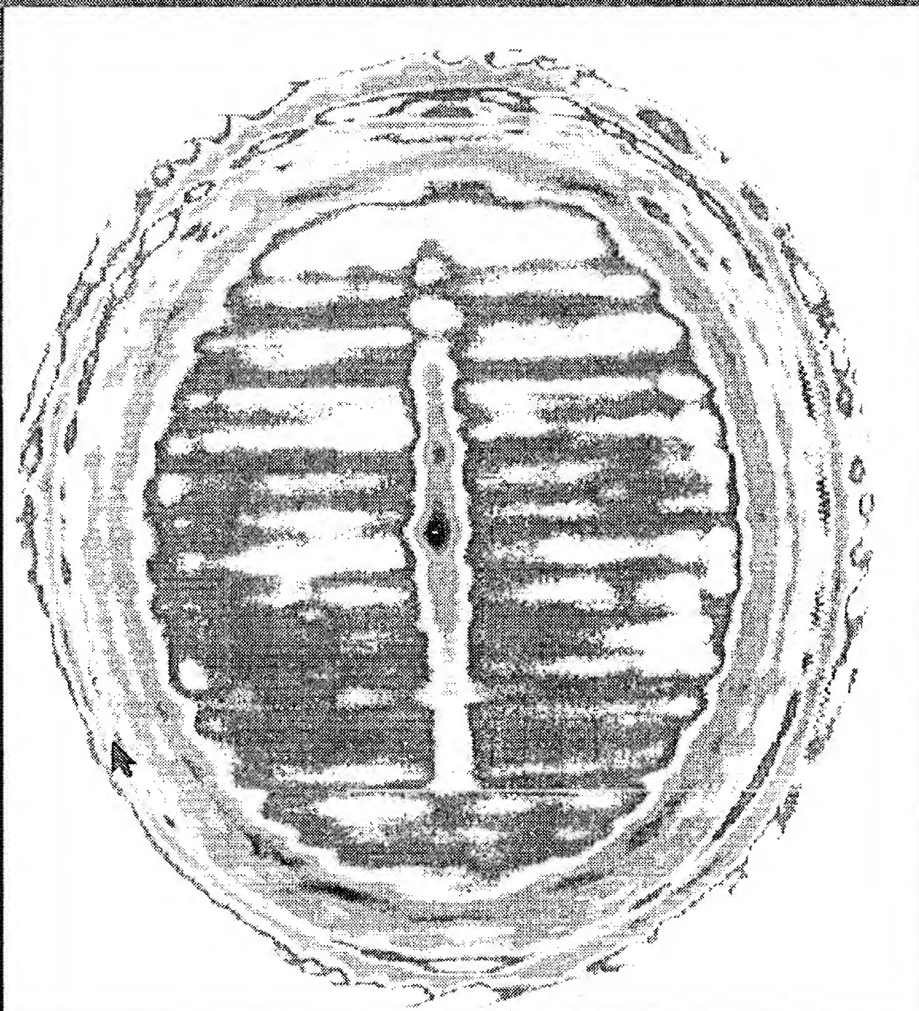
Spectrum Loading for Bonded Repairs

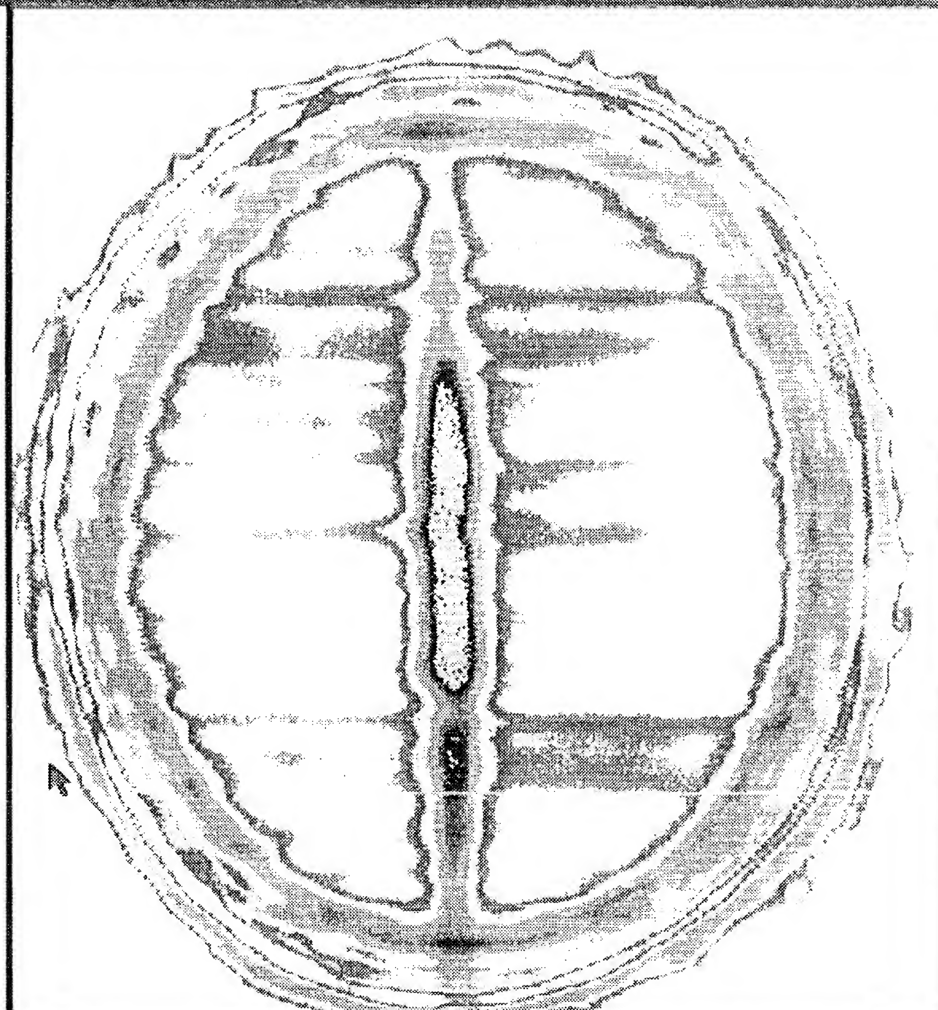
- Conservative spectrum for bonded repair?
- Effect of overloads under a bonded repair?
  - Plastic zones usually slow crack growth
  - Plastic zones under repair may be too small to retard crack growth
  - Increased debonding due to overloads may increase crack growth
- Investigate the effect of overloads first

Plastic Zone at Crack Tip



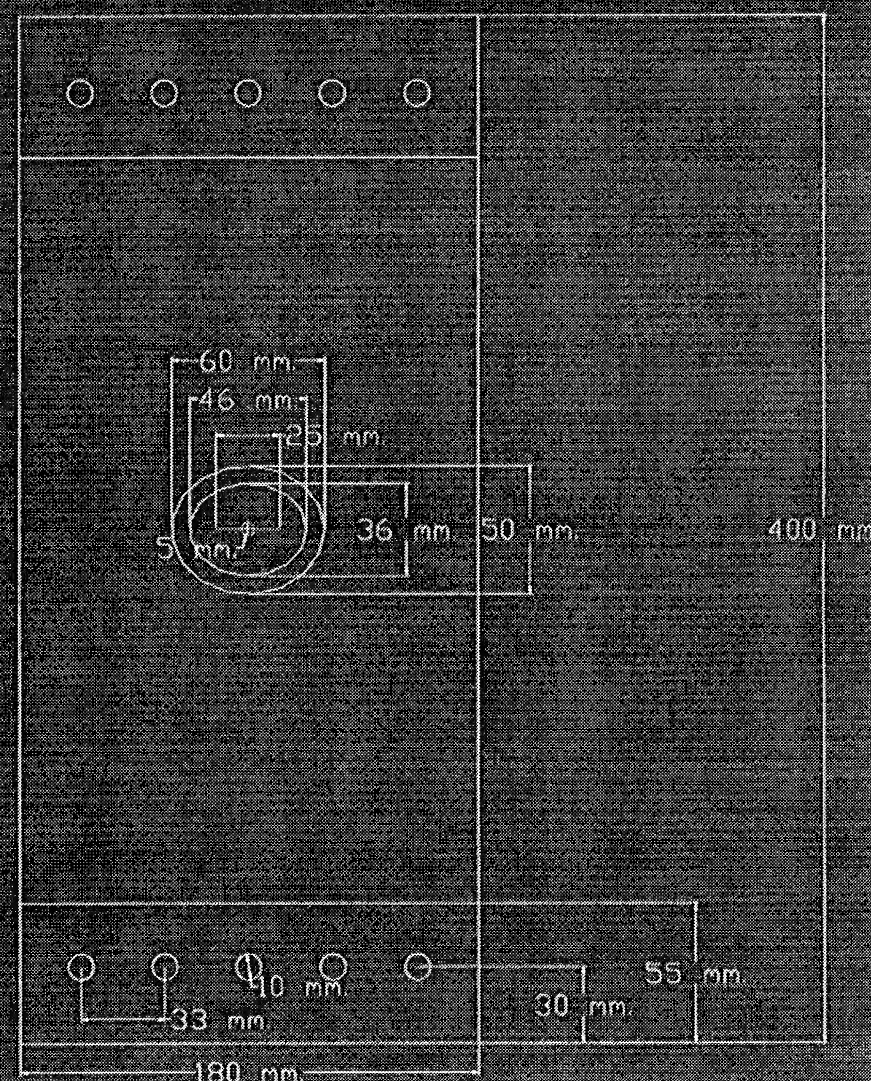
Debonding during crack growth

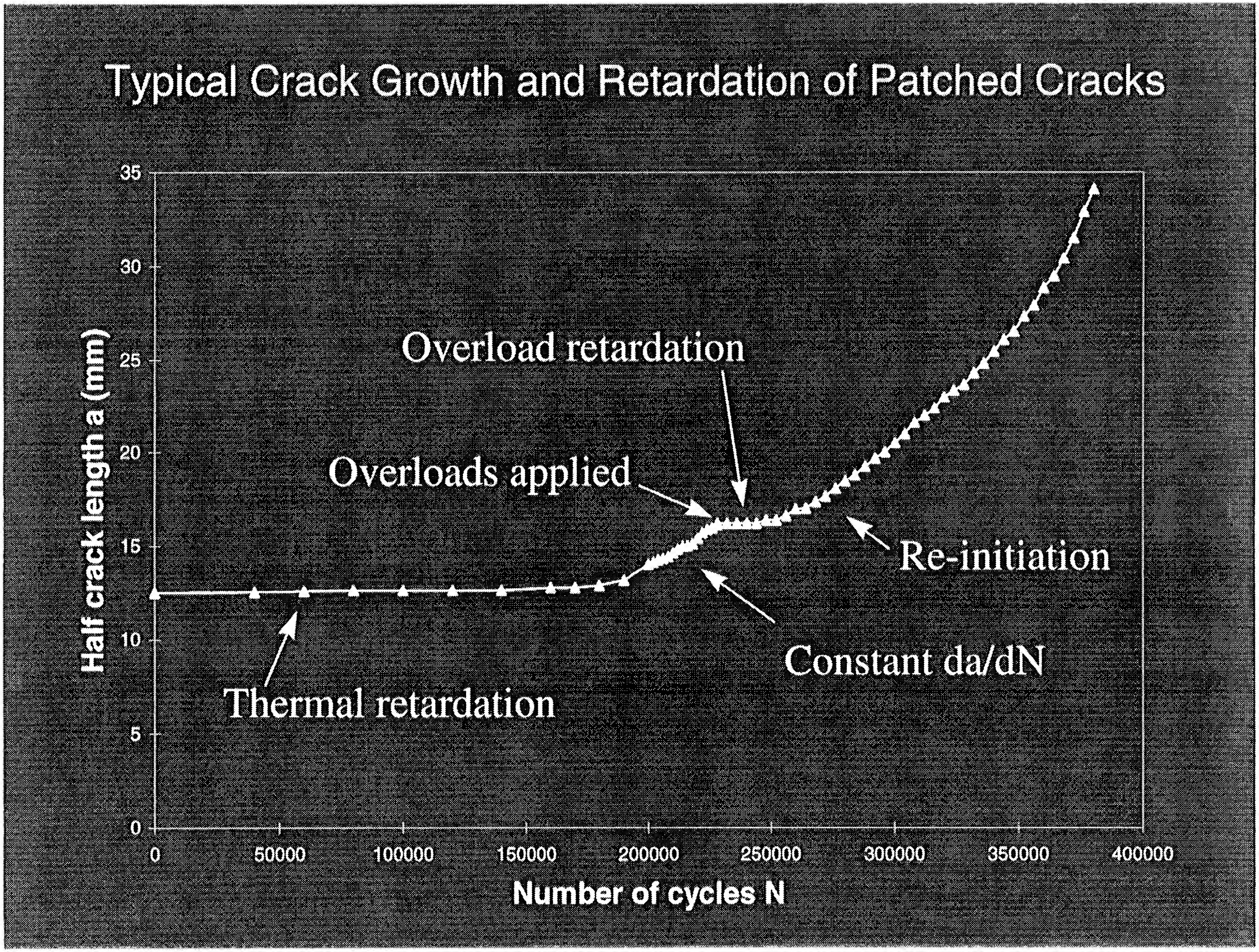




Overload and Spectrum Specimens

- 7075 panels
- Glare 2 3/2 patches
- Pre-cracking at lower stress
- Different sequenced overloads during constant amplitude loading
  - Normal loads: 6-120 MPa
  - 100 Peak loads: 6-200 MPa, @crack length of half patch width
- Measured crack growth rates
- C-scans to find evidence of disbonding




Typical Crack Growth and Retardation of Patched Cracks
Half crack length a (mm)
35
30
25
20
15
10
5
0
0
50000
100000
150000
200000
250000
300000
350000
400000
Number of cycles N
Overload retardation
Overloads applied
Re-initiation
Constant da/dN
Thermal retardation

Test Spectrum Development

- Based on Lockheed C-5A data
- Create integer cycles
- Clipping, Truncation and Filtering
- Testing on repaired cracks
- Spectrum verified on unpatched cracks

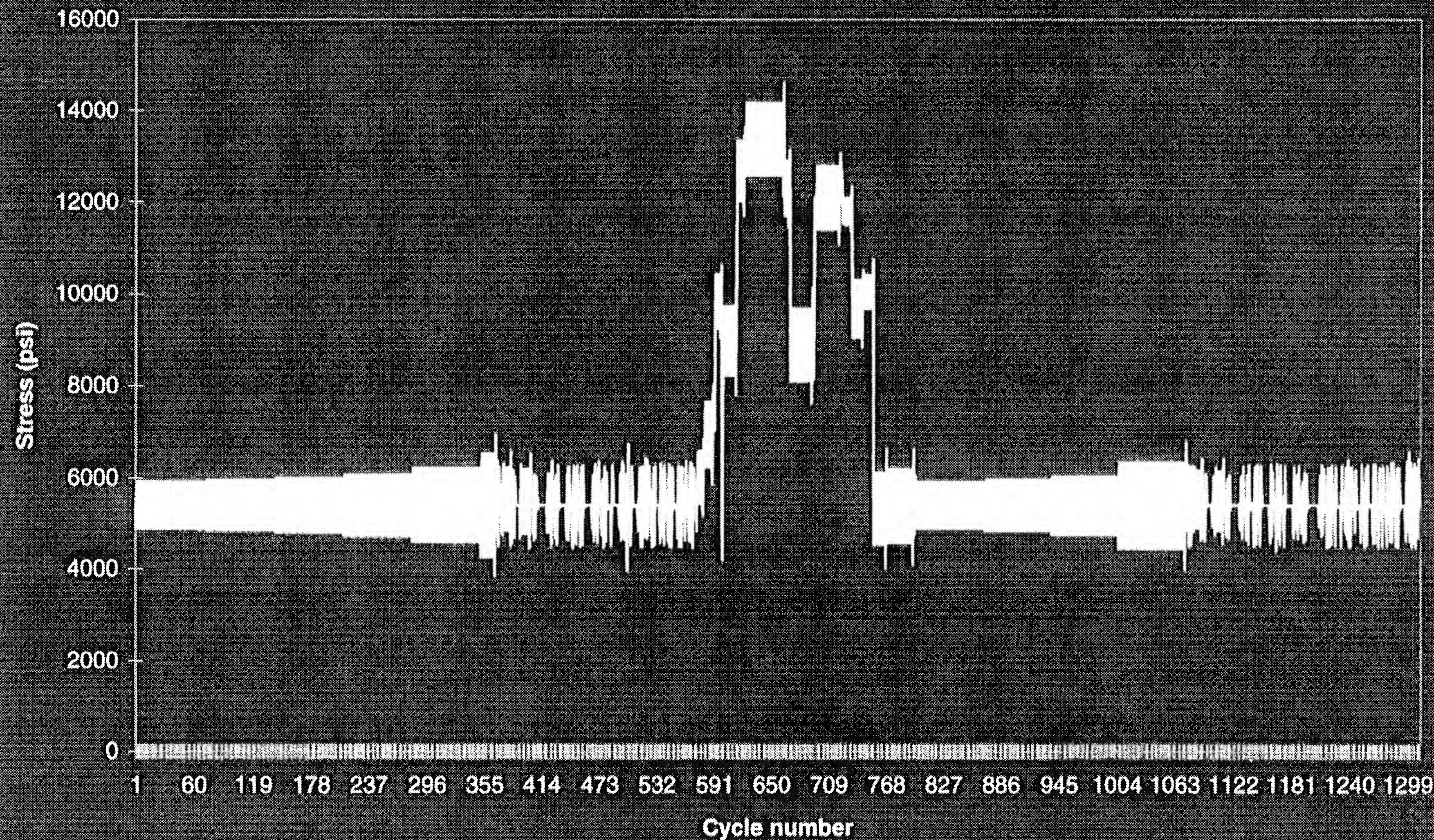
"Statistical" C-5A Spectrum Data

Occurences per flight	Minimum stress (psi)	Maximum stress (psi)
35.73743	4881.302	5916.822
7.501760	4263.112	6535.012
1.000000	3861.912	6936.212
0.4601240	3297.695	7500.429
3.8271800E-02	2547.69	8250.429
1.6042051E-03	1797.69	9000.429

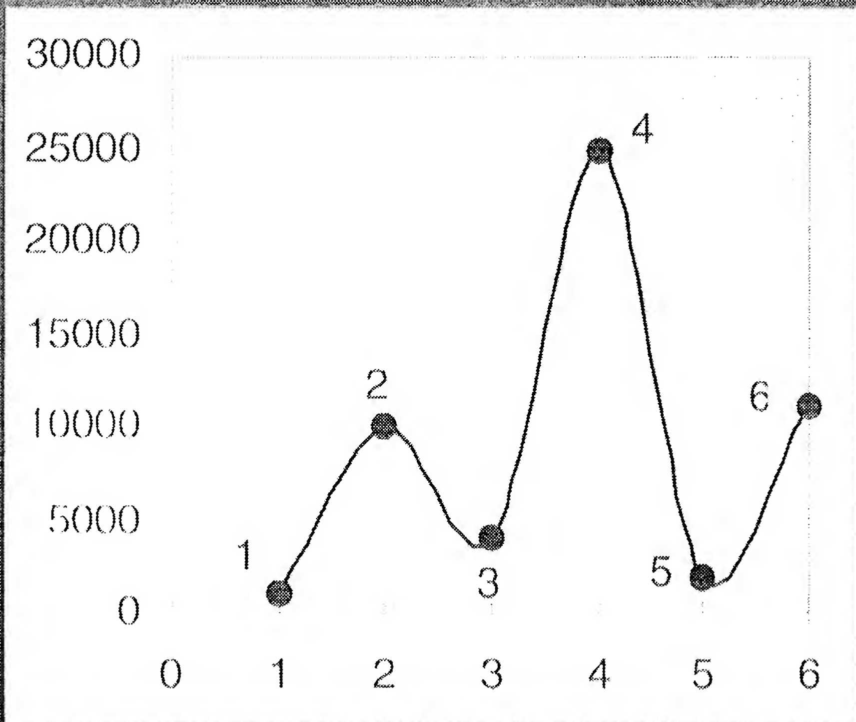
C-5A Spectrum Data

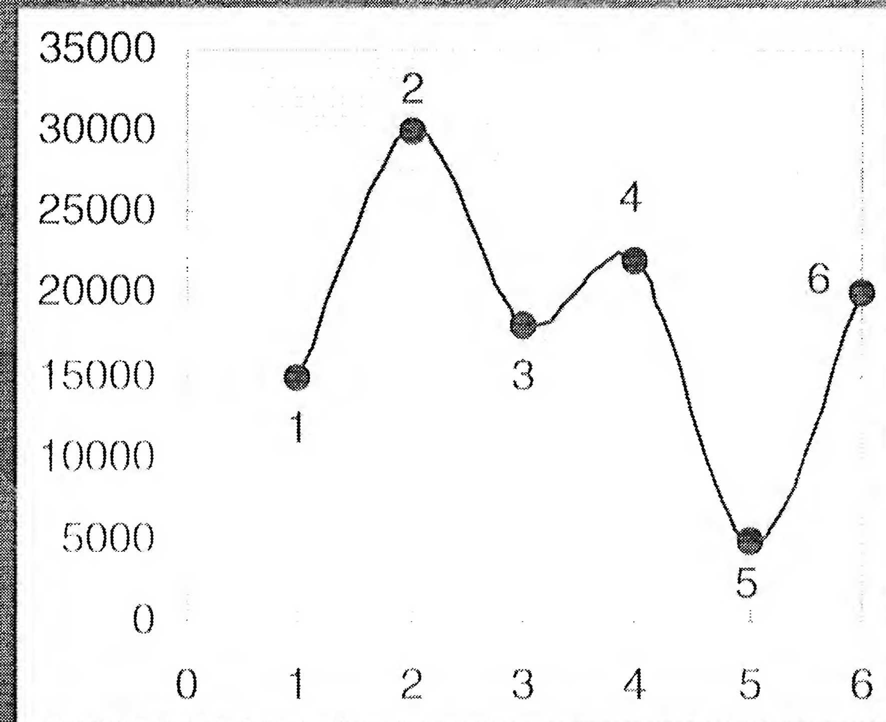
- 21 flight types,122 flights per block, 508 hrs.
- 60 blocks = 1 lifetime (~ 30,000 hrs.)
- Bucket filling creates integer cycles
- Bucket tolerance (fit of stress pairs)
- Max stress determines plastic zone size
- Retains correct R-ratio for crack growth

Flt01, no filter, 1300 cycles, test time 10 days



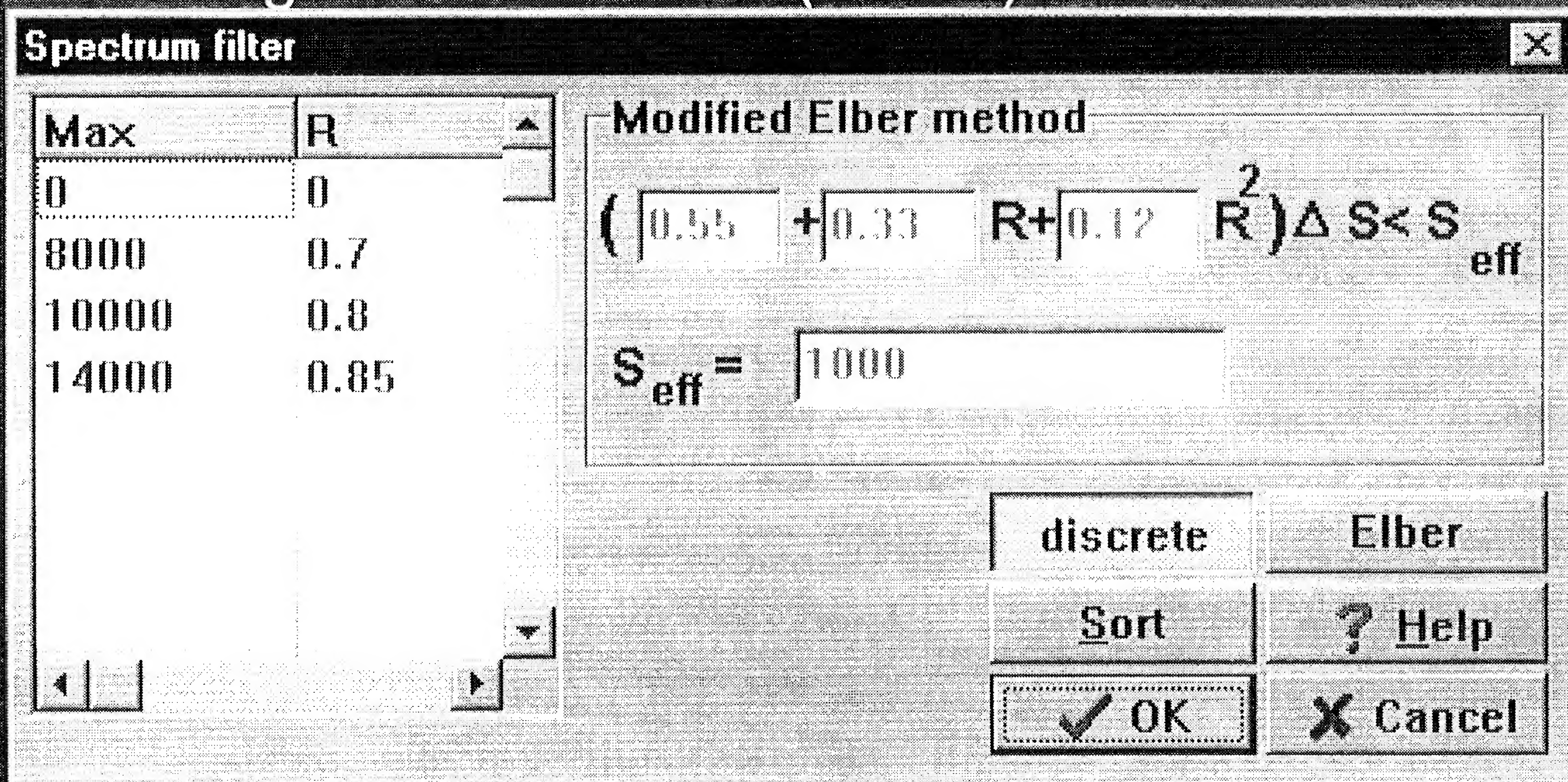
Rainflow cycle identification

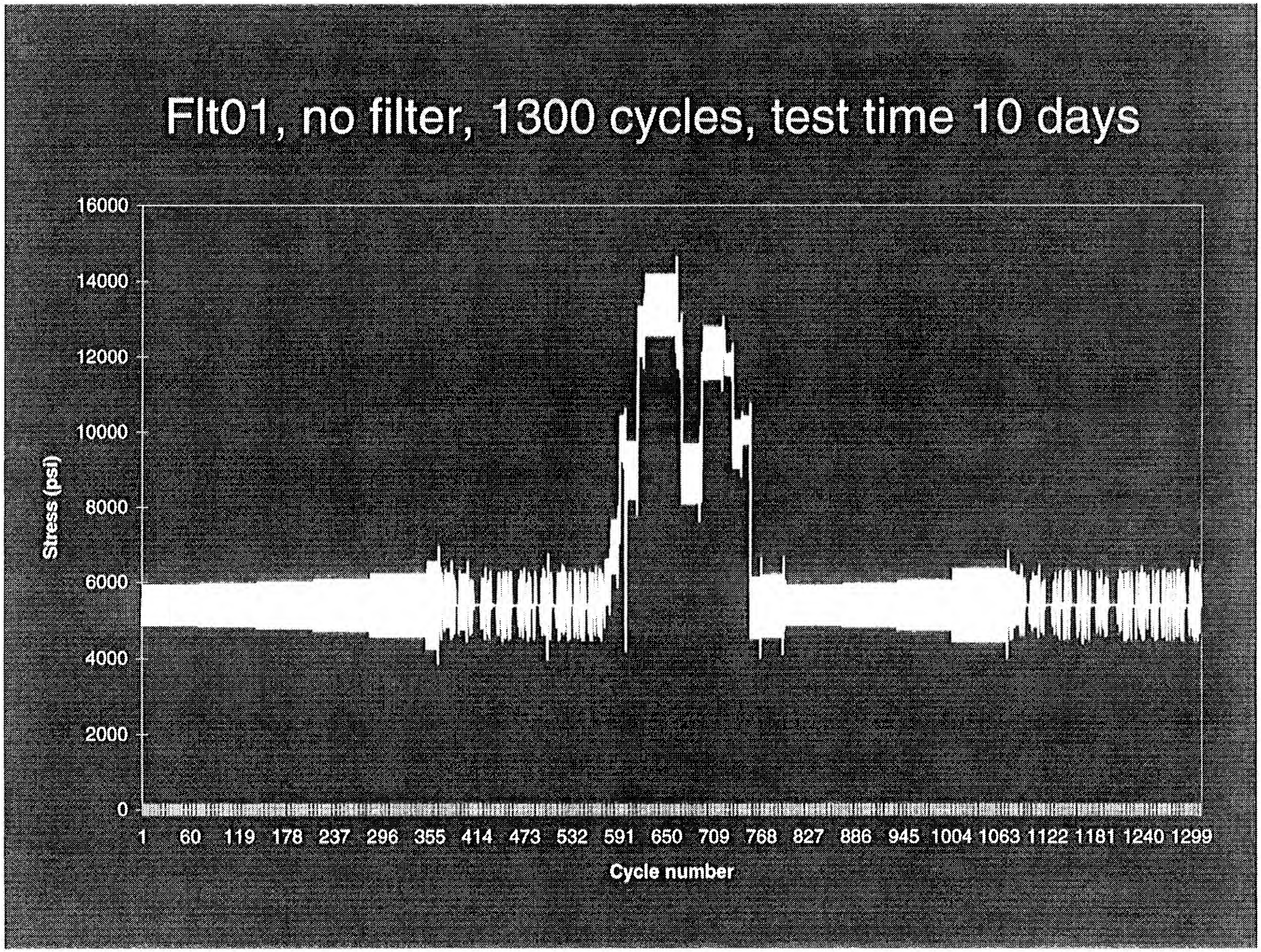




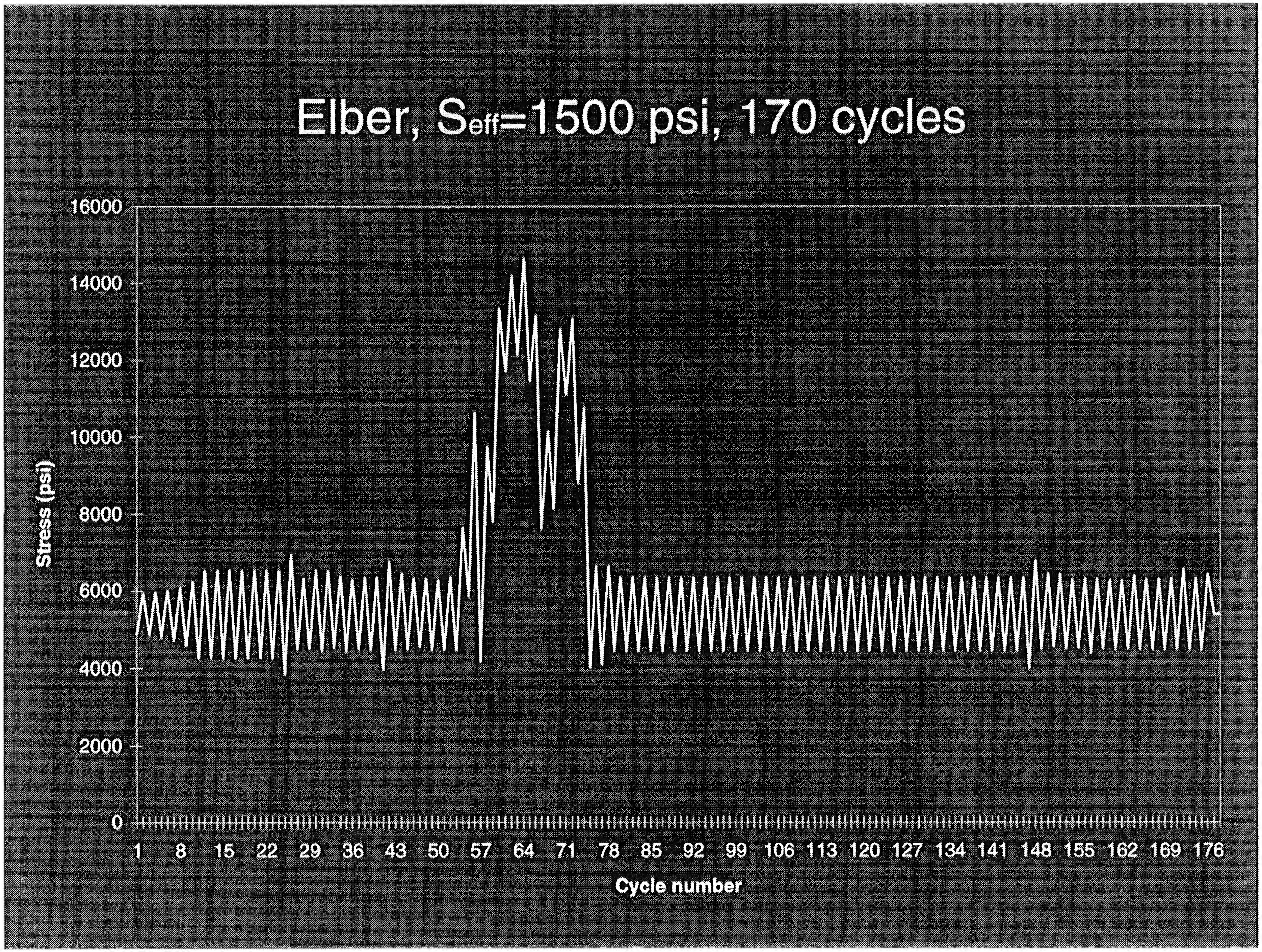
Data Reduction

- Rainflow cycle identification
- R-filter method and Elber filter method
- No truncation, <10/life do not occur
- Negative loads set at (almost) zero

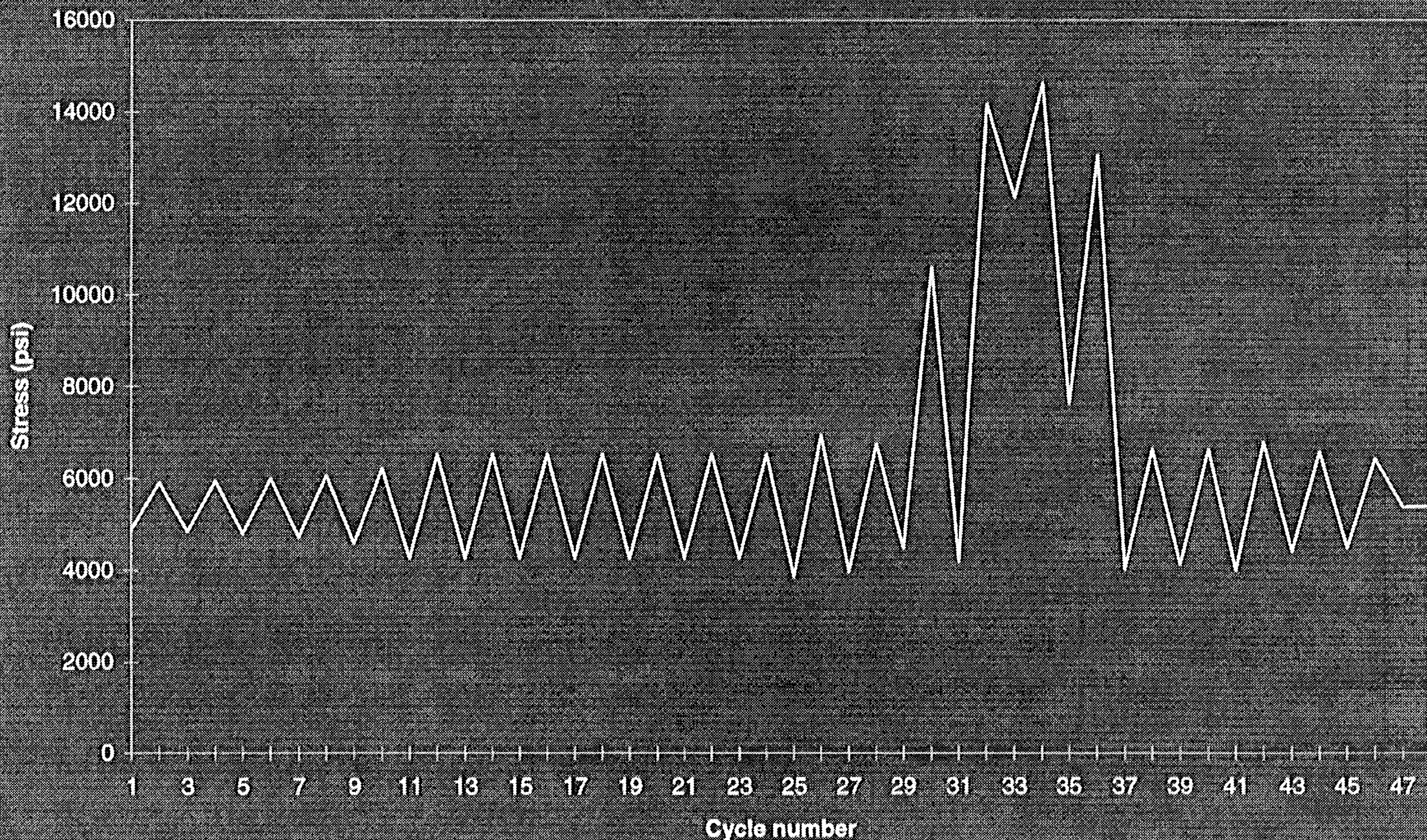


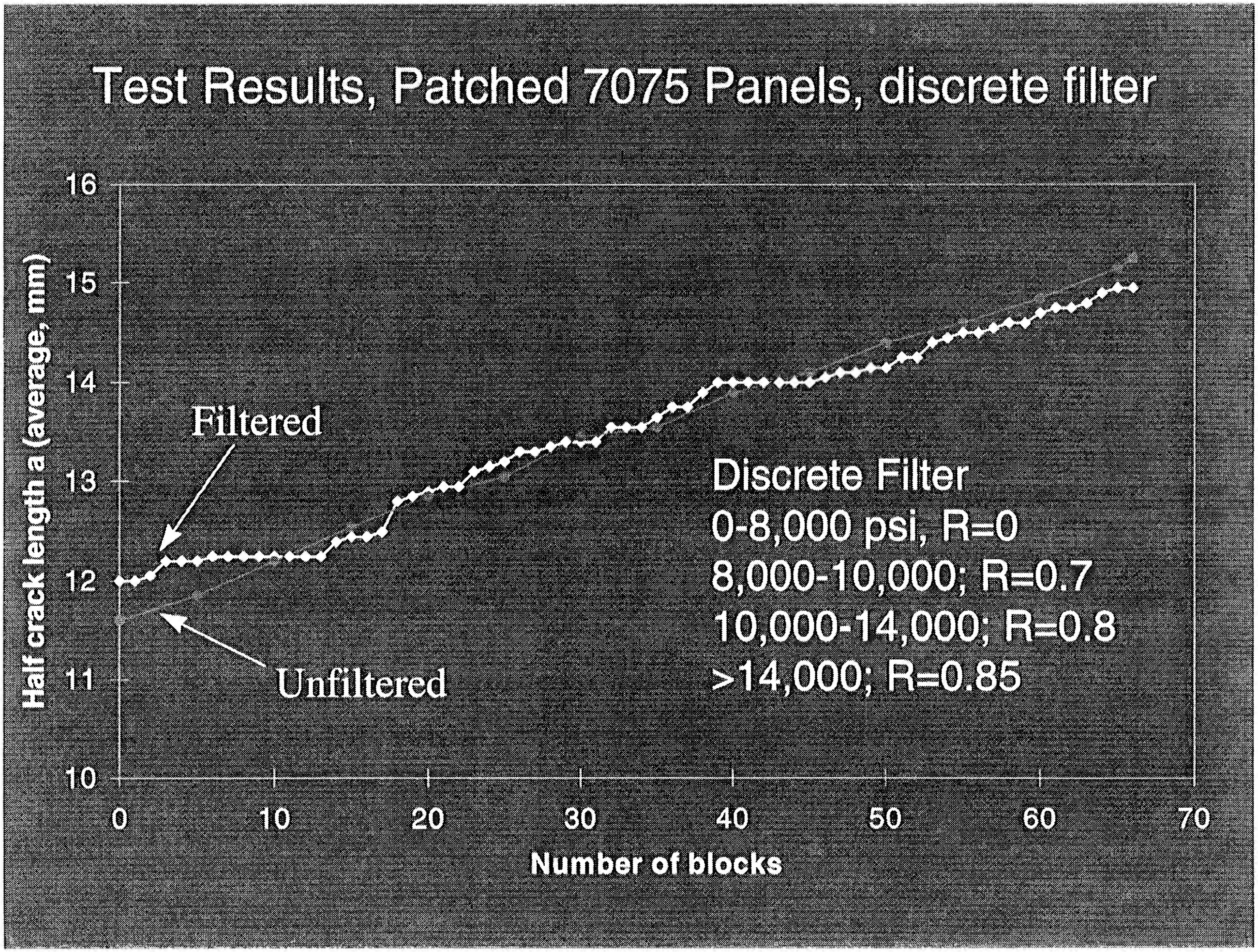


Flt01, no filter, 1300 cycles, test time 10 days
Stress (psi)
16000
14000
12000
10000
8000
6000
4000
2000
0
1 60 119 178 237 296 355 414 473 532 591 650 709 768 827 886 945 1004 1063 1122 1181 1240 1299
Cycle number

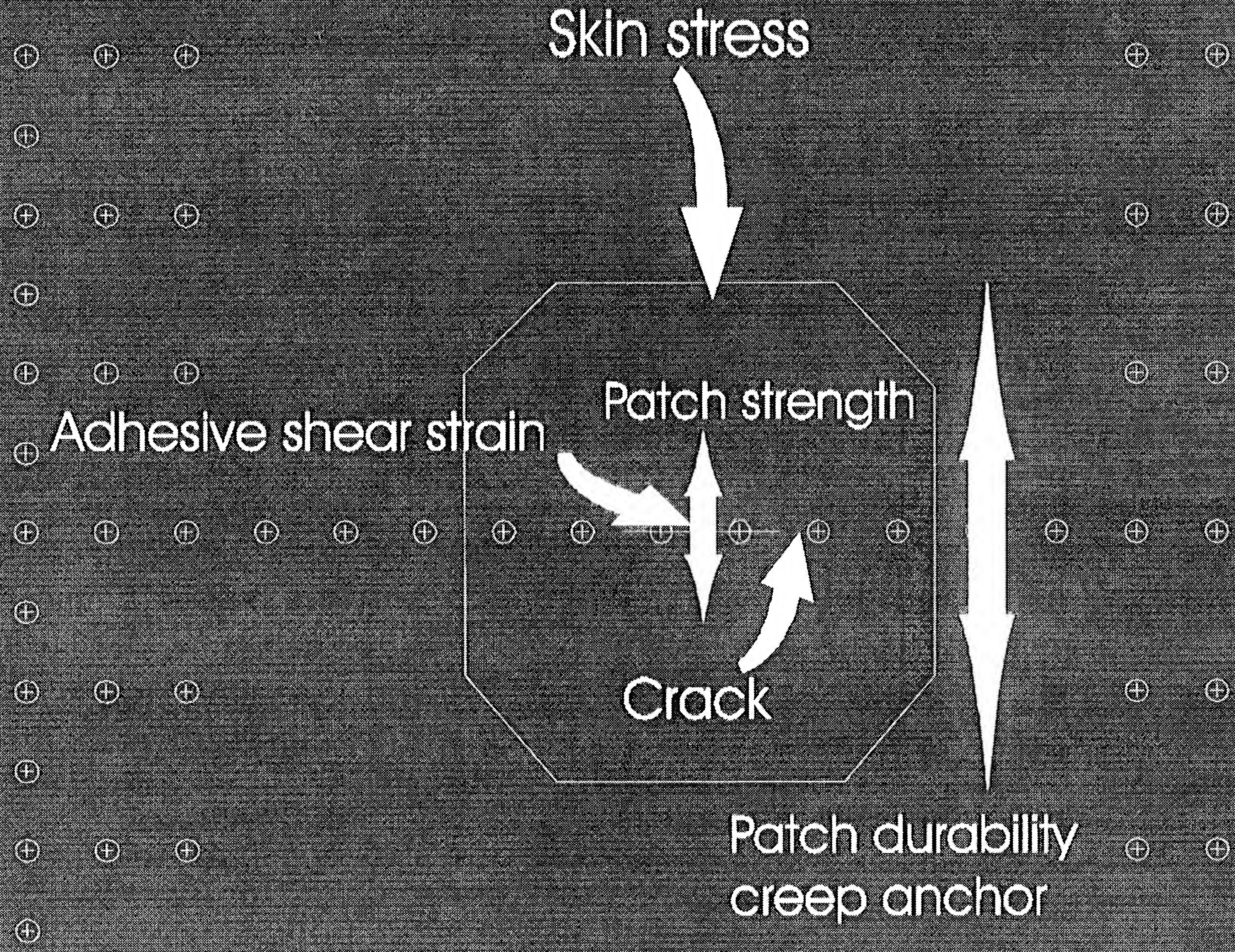

Elber, S_{eff}=1500 psi, 170 cycles
Stress (psi)
16000
14000
12000
10000
8000
6000
4000
2000
0
1
8
15
22
29
36
43
50
57
64
71
78
85
92
99
106
113
120
127
134
141
148
155
162
169
176
Cycle number

Elber, S_{eff}=1800 psi, 50 cycles, test time 1 day

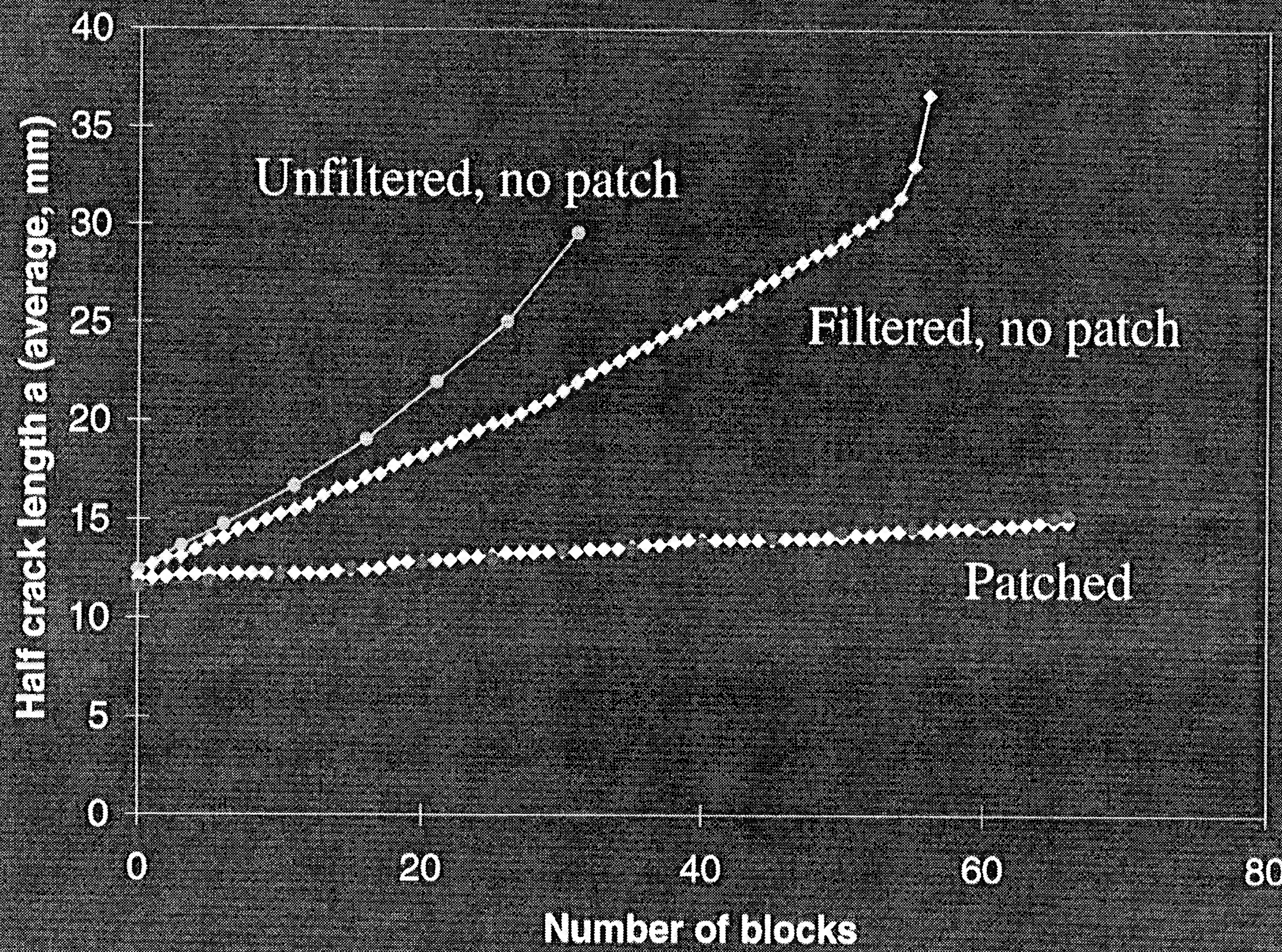



Test Results, Patched 7075 Panels, discrete filter
Half crack length a (average, mm)
16
15
14
13
12
11
10
0
10
20
30
40
50
60
70
Number of blocks
Filtered
Unfiltered
Discrete Filter
0-8,000 psi, R=0
8,000-10,000; R=0.7
10,000-14,000; R=0.8
>14,000; R=0.85

More Failure Modes than Crack Growth



Test Results, Unpatched panels



Conclusions

- Bonded repairs are effective in slowing crack growth under realistic crown fuselage spectra
- Effect of overloads on crack growth under patches is limited and corresponds with unpatched behavior
- Crack growth under bonded repairs is less sensitive to spectrum loading
- Multiple failure modes can make the usage of the correct spectrum vital

SESSION IV

BONDED COMPOSITE REPAIRS

Chairman -Major R. Fredell
United States Air Force Academy

Effects of bondline defects and environmental exposure on adhesively bonded composite patch repair of cracked aluminium alloy sheet

P. Poole, D.S. Lock and A. Young

Structural Materials Centre

Defence Evaluation and Research Agency

Farnborough, Hampshire, UK

DERA

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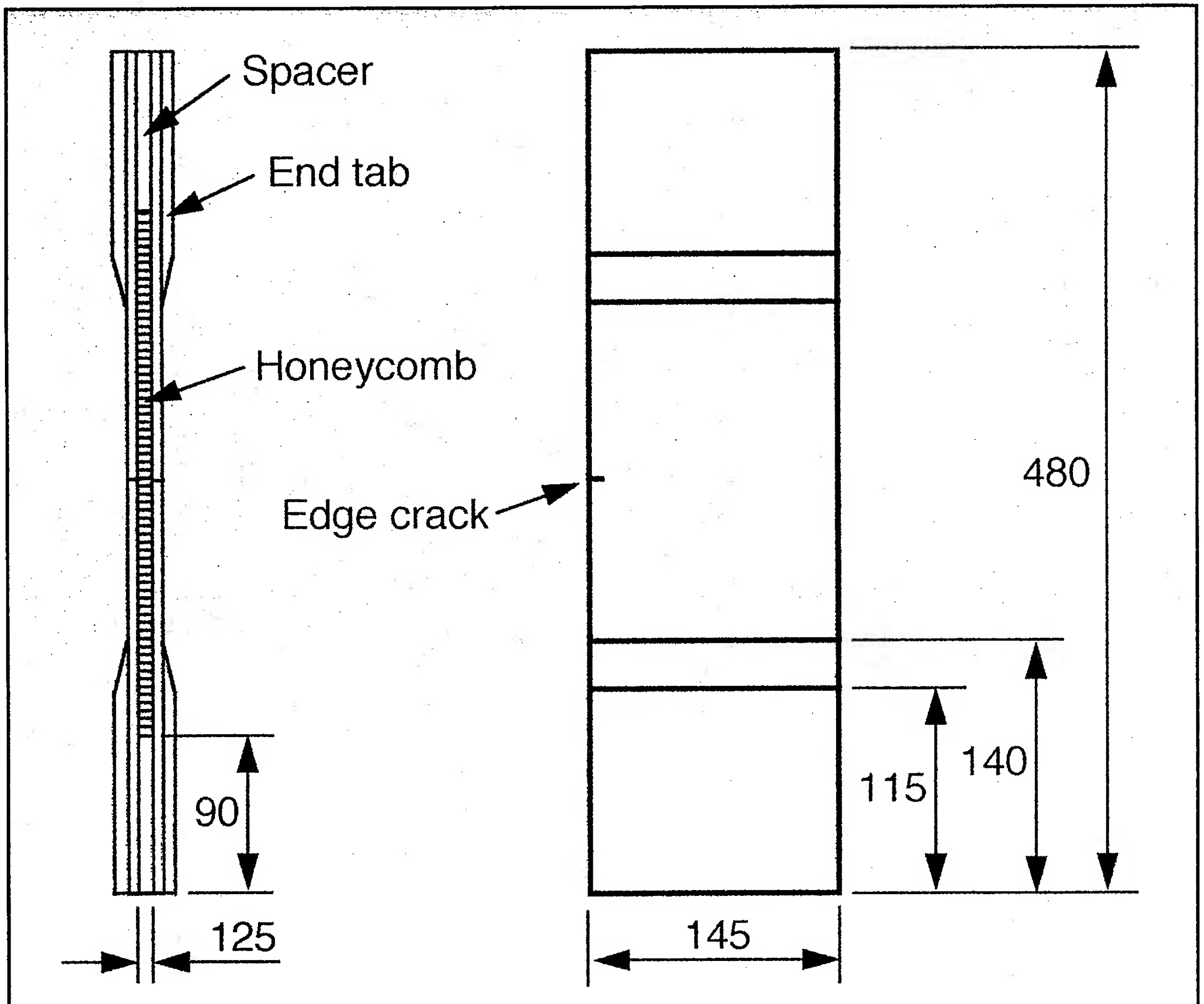
Dimensions of unpatched edge cracked specimen



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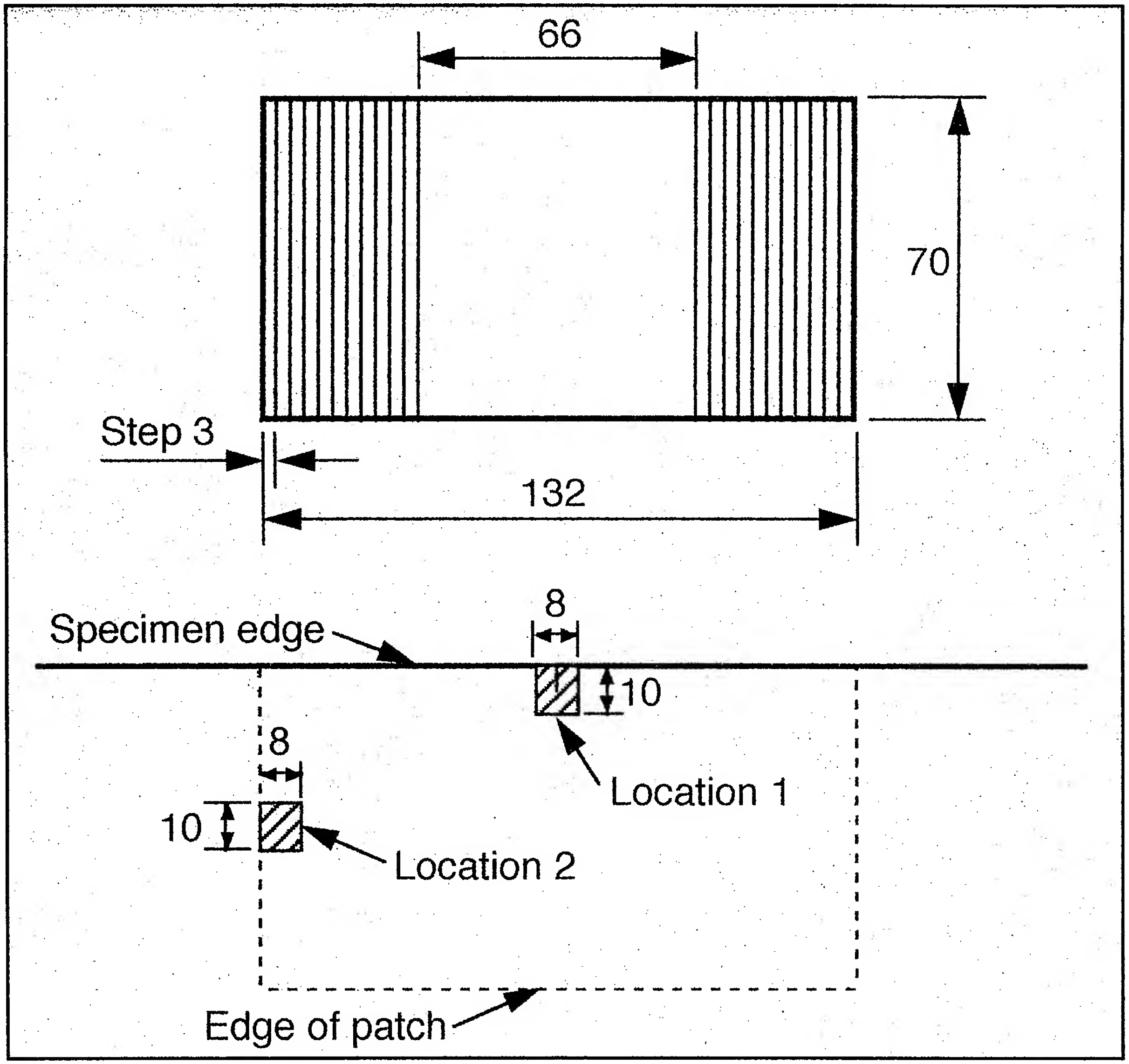
Dimensions and locations of patches and bondline defects



DERA

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Materials

- TL test panels
  - 4.0mm thick 2024-T3 aluminium alloy
  - initial crack length 7.5mm [load shedding to $S_{max} = 55MPa$, $R = 0.05$]
- BFRP patches
  - Textron 5521F/4; 12 plies
  - Avr thickness $t = 1.645mm$; $E = 208GPa$ [$Et = 342$]
- CFRP patches
  - Ciba - Geigy T800/924; 16 plies
  - Avr thickness $t = 2.00mm$; $E = 168GPa$ [$Et = 336$]
- Adhesive - Redux 312/5

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DERA

Experimental

- Adhesive bonding
  - grit blast/silane swab; Teflon inserts
  - no corrosion inhibiting primer
  - Redux 312/5 [120°C cure in autoclave]
- Fatigue testing
  - $R = 0.05$; $S_{MAX} = 110MPa$; 5Hz
  - Crack length measurement - eddy current and plastic replica techniques
  - Debonding - hand held ultrasonic probe
- Environmental exposure - water at 74°C for up to 62 days

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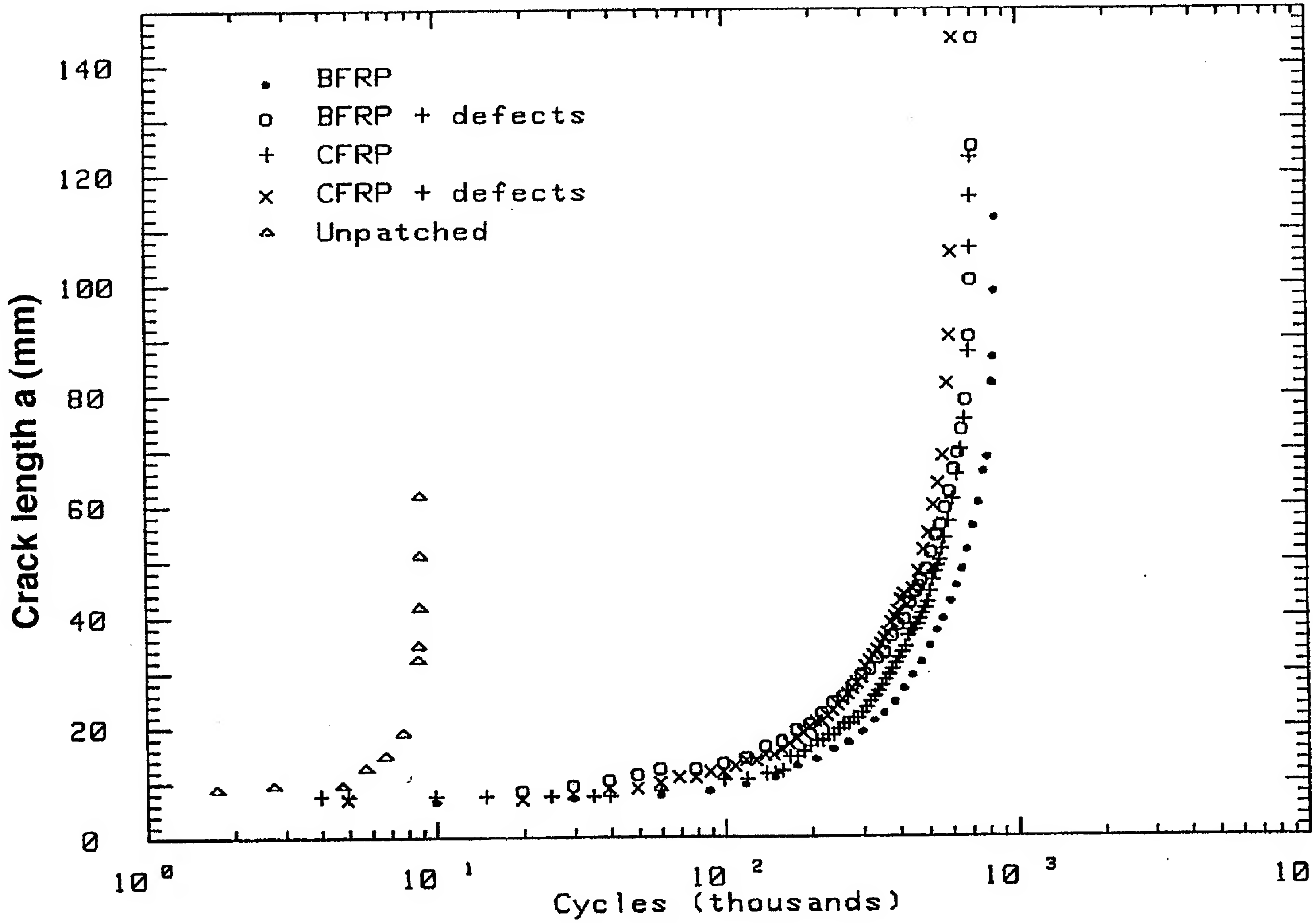
Effect of patch type and defects on growth of fatigue cracks



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Effect of patch type and defects on fatigue life

Patch type	Defects	Life,cycles
Unpatched		8959
BFRP	No	858745
BFRP	Yes	722400
CFRP	No	704641
CFRP	Yes	622732

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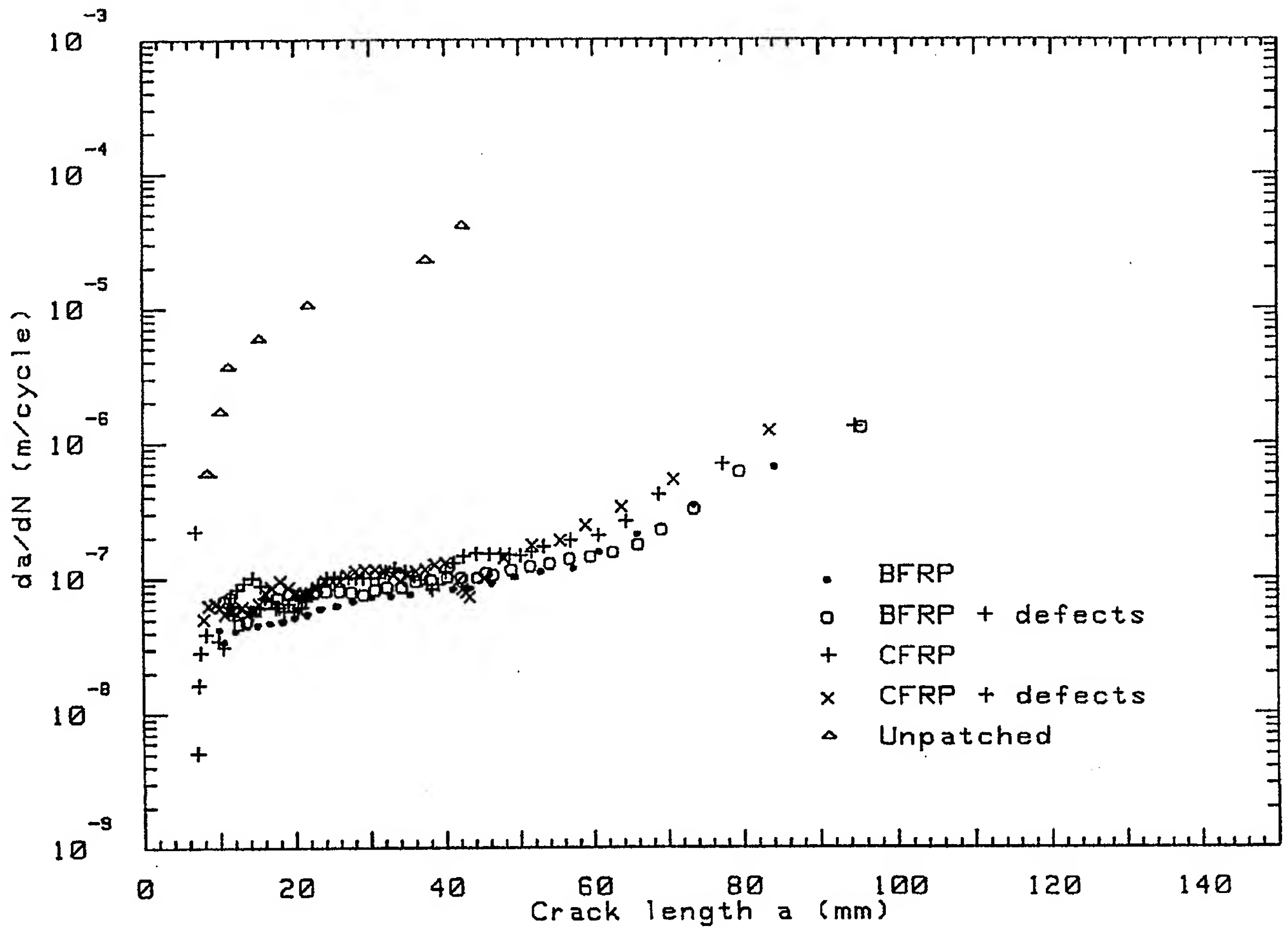
Effect of patch type and defects on rate of crack growth



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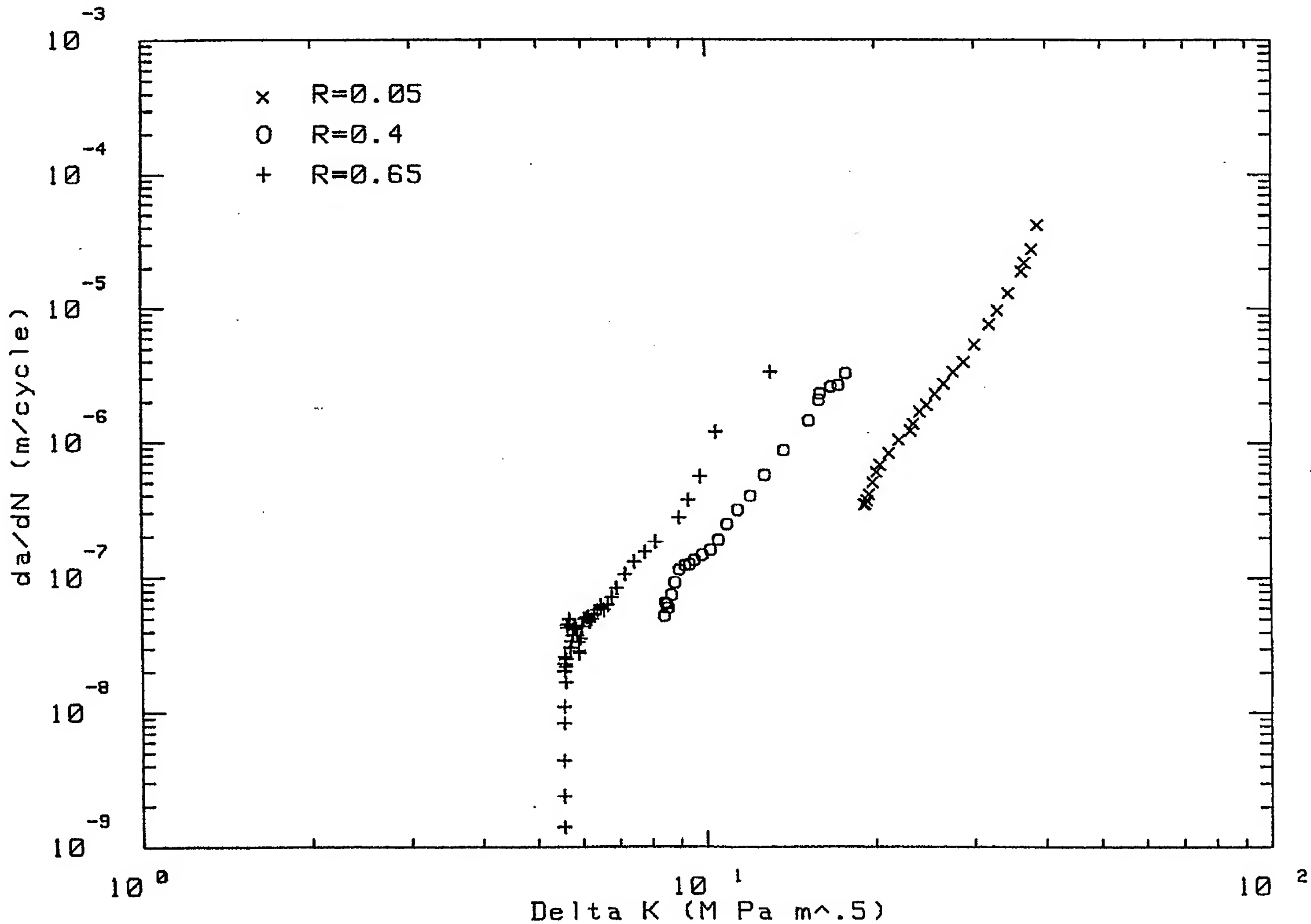
Effect of R-ratio on rate of fatigue crack growth



DERA

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Values of $\Delta K^p/\Delta K^u$ determined from fatigue crack growth rate

Patch type	a,mm	R	da/dN, m/c	ΔK^p	ΔK^u	$\Delta K^p/\Delta K^u$
BFRP	15	0.59	4.8×10^{-8}	6.3	25.3	0.25
	30	0.57	7.5×10^{-8}	7.1	35.1	0.20
	60	0.49	1.4×10^{-7}	8.7	50.3	0.17
CFRP + defects	15	0.64	8.0×10^{-8}	6.9	25.3	0.27
	30	0.62	1.1×10^{-7}	7.3	35.1	0.21
	60	0.53	2.3×10^{-7}	9.7	50.3	0.19

DERA

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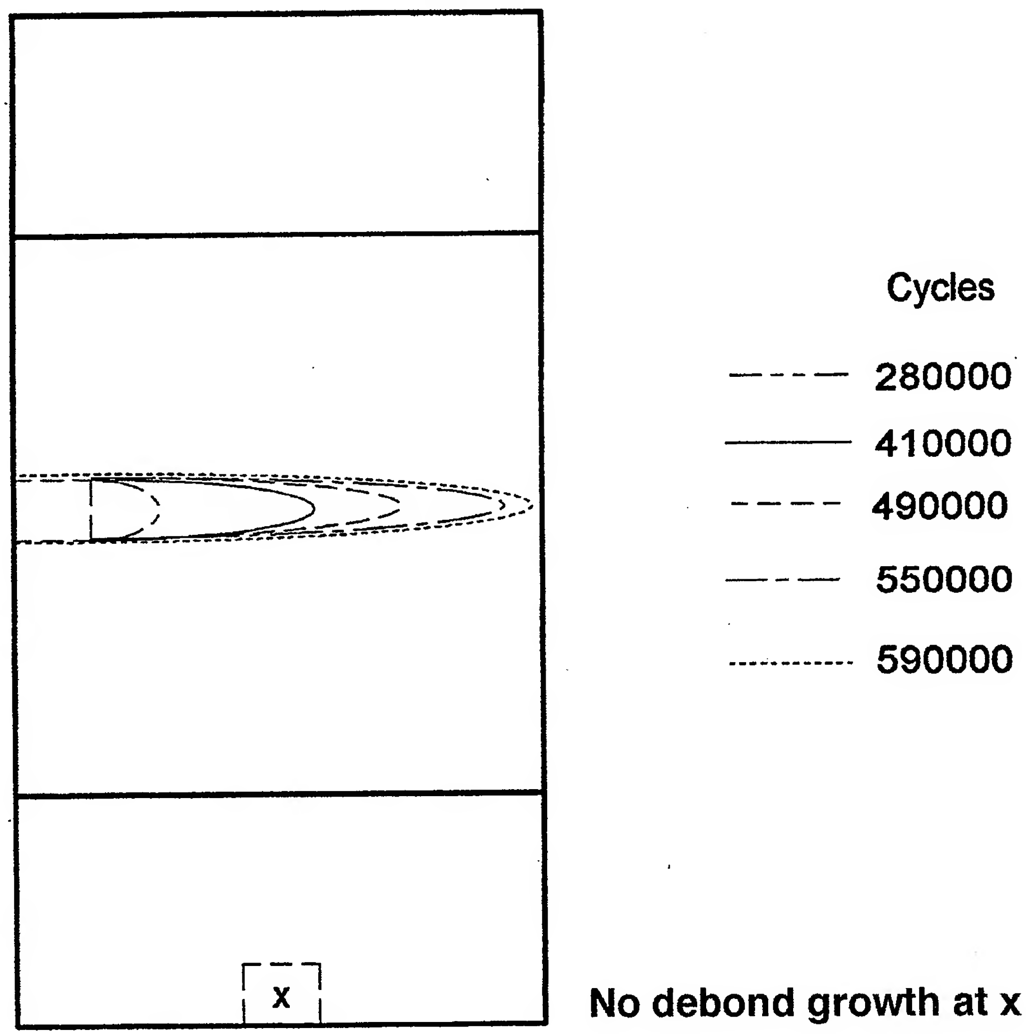
Growth of debond from bondline defect in CFRP patched specimen



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Numerical Analysis

- Coupled BE/FE computer model for 3-D analysis
- Cracked metal sheet - modelled using BEM
- Composite patches - modelled using FEM
- Adhesive - modelled using spring elements
- Assumed material properties
  - 2024: $t = 4.0$mm, $E = 71$GPa, $\alpha = 23 \times 10^{-6}$ $[^\circ C]^{-1}$
  - BFRP: $t = 1.645$mm, $E = 208$GPa, $\alpha = 4.5 \times 10^{-6}$ $[^\circ C]^{-1}$
  - CFRP: $t = 2.0$mm, $E = 168$GPa, $\alpha = 0.4 \times 10^{-6}$ $[^\circ C]^{-1}$
  - 312/5: $t = 0.15$mm, $G = 0.8$GPa
- Loading assumed to be applied by rigid grips
- Debonding modelled assuming idealised geometry

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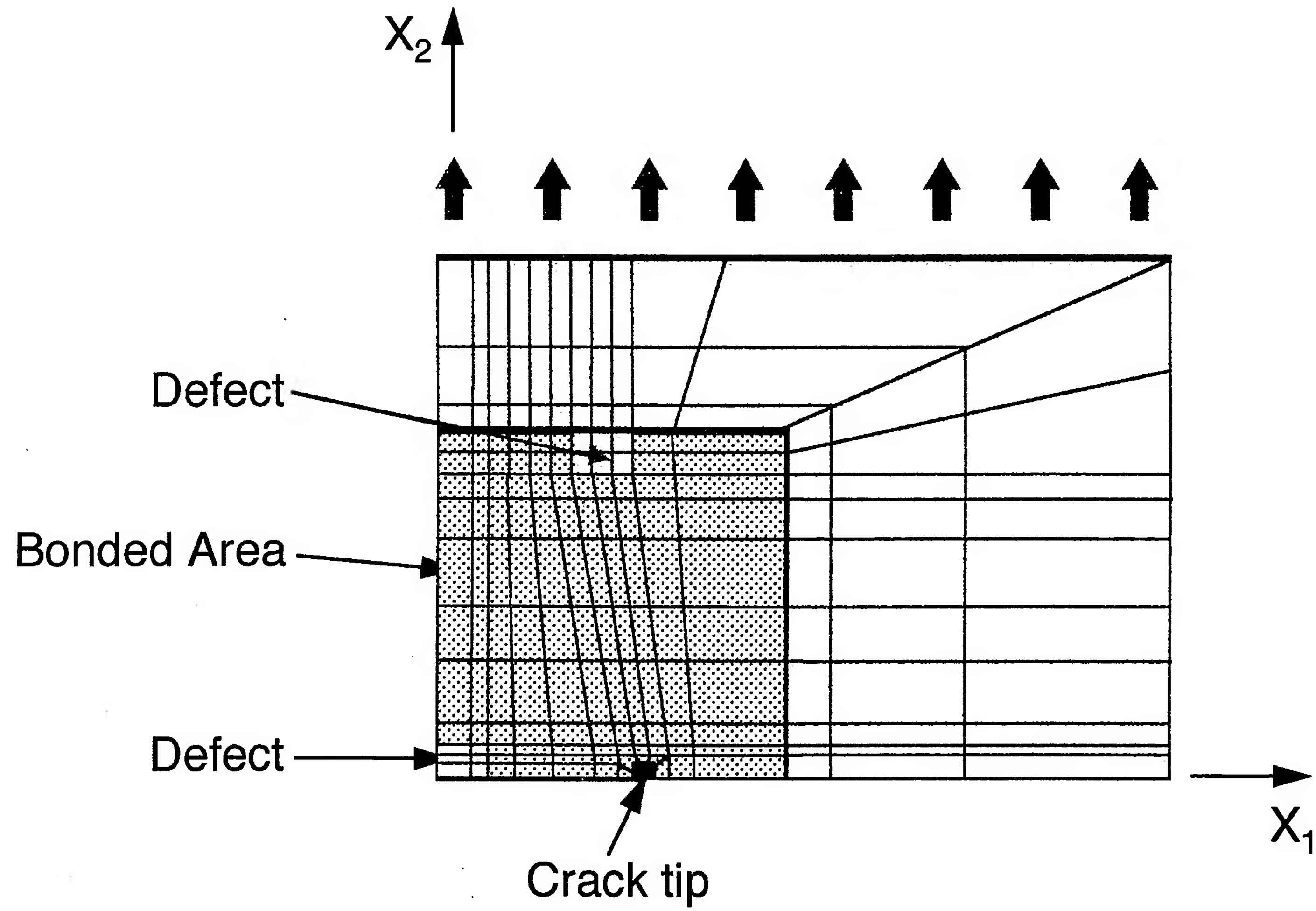
Element mesh used in BE/FE patch repair analysis



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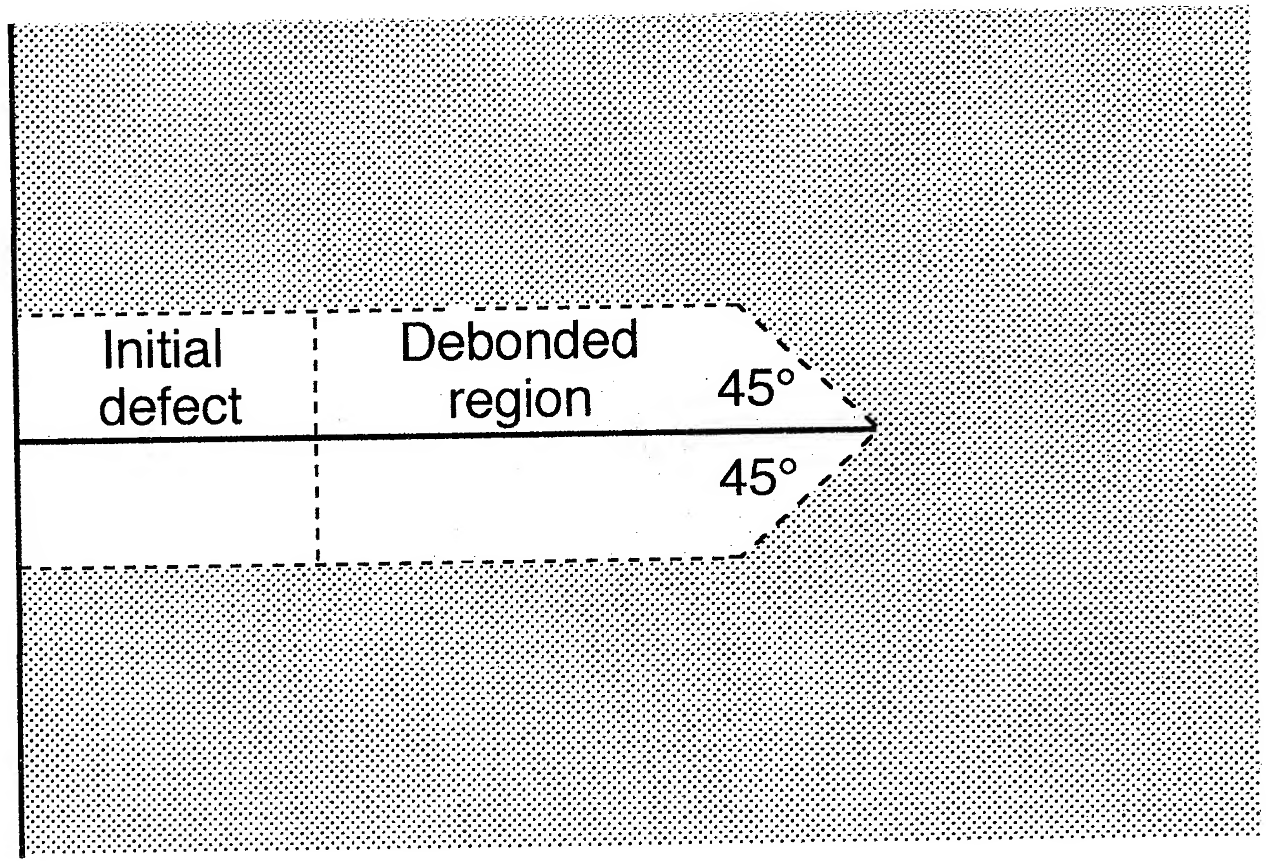
Shape and size of debond region assumed in theoretical study



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Stress intensity factors due to residual thermal stress only

	K^{therm} (MPa.m$^{1/2}$)				
	a = 7.5mm	*a* =15mm	*a* = 30mm	*a* = 60mm	*a* = 90mm
BFRP	8.5	8.6	8.2	6.5	6.9
BFRP/defects	9.9	9.4	8.4	6.5	6.8
CFRP	10.3	10.5	10.0	7.9	8.3
CFRP/defects	12.5	11.4	10.2	7.9	8.3

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Stress intensity factor for unconstrained residual thermal stresses and loading applied by rigid displacement constraints

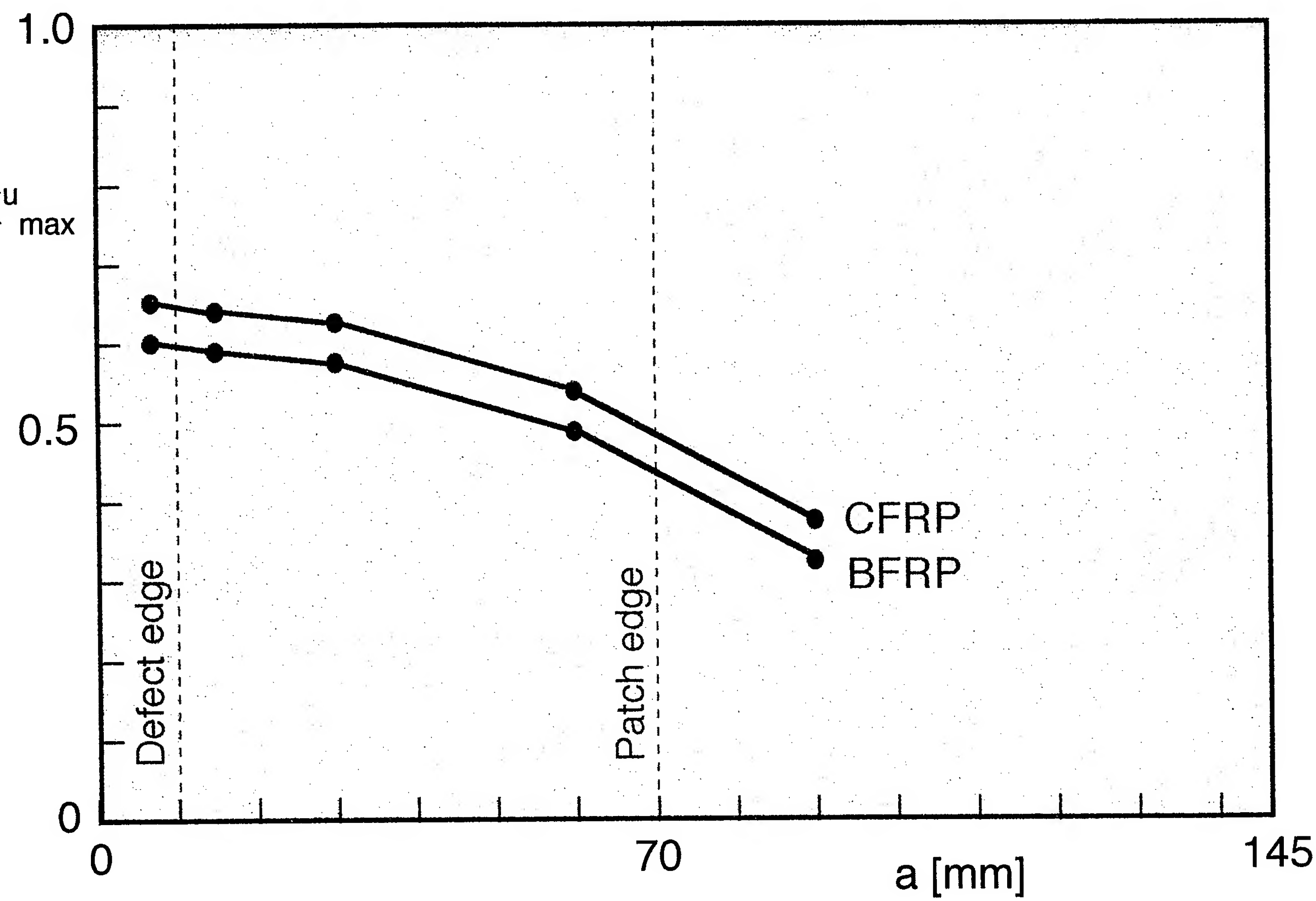
K_{min} and K_{max} [MPa.m$^{1/2}$]					
	a = 7.5mm	a = 15mm	a = 30mm	a = 60mm	a =90mm
unpatched	0.9 - 18.9	1.3 - 26.7	1.8 - 37.0	2.6 - 53.0	3.8 - 76.6
BFRP	8.8 - 14.6	8.9 - 15.0	8.5 - 14.8	6.9 - 14.2	7.8 - 24.2
BFRP/defects	10.2 - 17.2	9.8 - 16.3	8.7 - 15.1	6.9 - 14.2	7.7 - 24.1
CFRP	10.6 - 16.4	10.8 - 16.9	10.3 - 16.7	8.3 - 15.6	9.2 - 25.7
CFRP/defects	12.9 - 19.9	11.8 - 18.4	10.5 - 16.9	8.3 - 15.6	9.2 - 25.8

ΔK [MPa.m$^{1/2}$] ($R = K_{min} / K_{max}$)					
	a = 7.5mm	a 15mm	a = 30mm	a = 60mm	a =90mm
unpatched	17.9 (0.05)	25.3 (0.05)	35.1 (0.05)	50.3 (0.05)	72.8 (0.05)
BFRP	5.8 (0.60)	6.1 (0.59)	6.3 (0.57)	7.3 (0.49)	16.4 (0.32)
BFRP/defects	7.0 (0.59)	6.6 (0.60)	6.4 (0.58)	7.3 (0.49)	16.4 (0.32)
CFRP	5.8 (0.65)	6.1 (0.64)	6.4 (0.62)	7.3 (0.53)	16.5 (0.36)
CFRP/defects	7.0 (0.65)	6.6 (0.64)	6.4 (0.62)	7.3 (0.53)	16.6 (0.36)

$\Delta K^p / \Delta K^u$					
	a = 7.5mm	a = 15mm	a = 30mm	a = 60mm	a =90mm
BFRP	0.32	0.24	0.18	0.15	0.23
BFRP/defects	0.39	0.26	0.18	0.15	0.23
CFRP	0.32	0.24	0.18	0.15	0.23
CFRP/defects	0.39	0.26	0.18	0.15	0.23

DERA

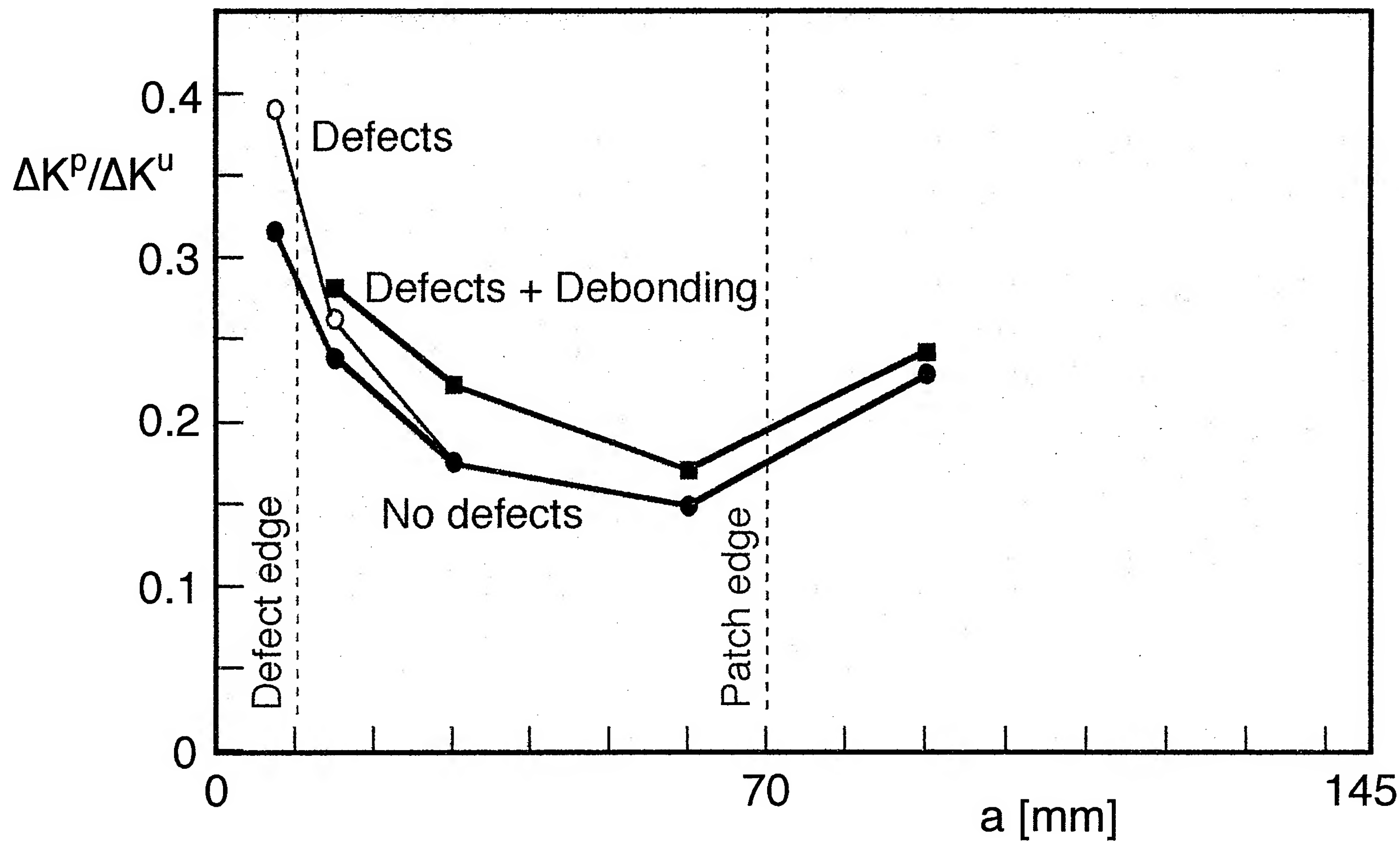
Variation of $R = K_{min} / K_{max}$ with crack length



DERA

8019707350/BL/POOLE

Variation $\Delta K^p/\Delta K^u$ with crack length



DERA

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Comparison of values of $\Delta K^p/\Delta K^u$ determined from experimental crack growth rate data with those predicted by analytical model

Patch type	a,mm	$\Delta K^p / \Delta K^u$ (Expt)	$\Delta K^p / \Delta K^u$ (Theoretical)	
			No defect/ debond	Defect & debond
BFRP	15	0.25	0.24	0.28
	30	0.20	0.18	0.22
	60	0.17	0.15	0.17
CFRP + defects	15	0.27	0.24	0.28
	30	0.21	0.18	0.22
	60	0.19	0.15	0.17

DERA

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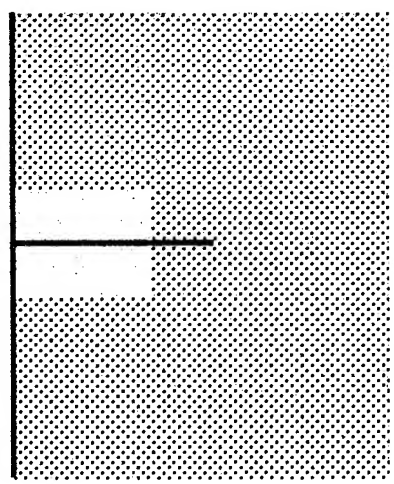
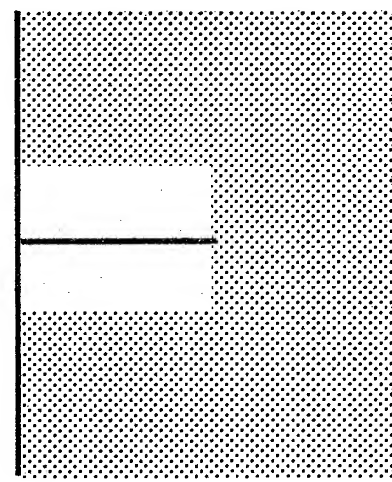
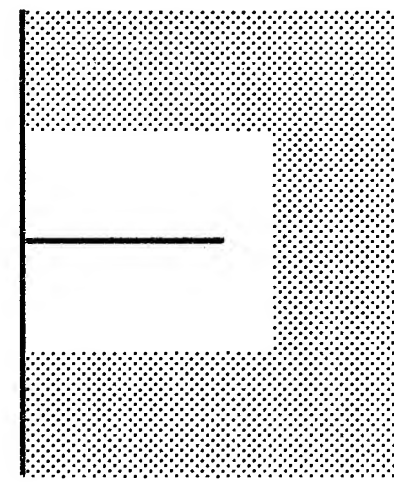
Defect sizes assumed in theoretical study



Defect 10mm x 8mm
(a = 15mm)



Defect 15mm x 12mm
(a = 15mm)



Defect 20mm x 16mm
(a = 15mm)

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Effect of initial defect size (crack length a =15mm)

K_{min} and K_{max} [MPa.m$^{1/2}$]				
	no defect	10mm x 8mm	15mm x 12mm	20mm x 16mm
unpatched	1.3 - 26.7	1.3 - 26.7	1.3 - 26.7	1.3 - 26.7
BFRP patch	8.9 - 15.0	9.8 - 16.3	12.1 - 20.2	13.2 - 21.9
CFRP PATCH	10.8 - 16.9	11.8 - 18.4	14.4 - 22.6	15.7 - 24.5

ΔK [MPa.m$^{1/2}$] ($R = K_{min}/K_{max}$)				
	no defect	10mm x 8mm	15mm x 12mm	20mm x 16mm
unpatched	25.4 (0.05)	25.4 (0.05)	25.4 (0.05)	25.4 (0.05)
BFRP patch	6.1 (0.59)	6.6 (0.60)	8.1 (0.60)	8.8 (0.60)
CFRP patch	6.1 (0.64)	6.6 (0.64)	8.1 (0.64)	8.8 (0.64)

$\Delta K^p/\Delta K^u$				
	no defects	10mm x 8mm	15mm x 12mm	20mm x 16mm
BFRP patch	0.24	0.26	0.32	0.35
CFRP patch	0.24	0.26	0.32	0.35

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Possible effects of exposure to hot-wet environments

- Water absorption by composite patch
- Water absorption by adhesive
  - effect on shear stress - shear strain behaviour [eg reduced modulus]
- Loss of adhesion at aluminium/adhesive interface
- Corrosion of aluminium alloy
  - crevice corrosion [edge defects, adhesive spew]
  - galvanic corrosion [CFRP/Al; clad layer]
  - corrosion product wedging
  - protection schemes [eg woven GRP ply next to Al]

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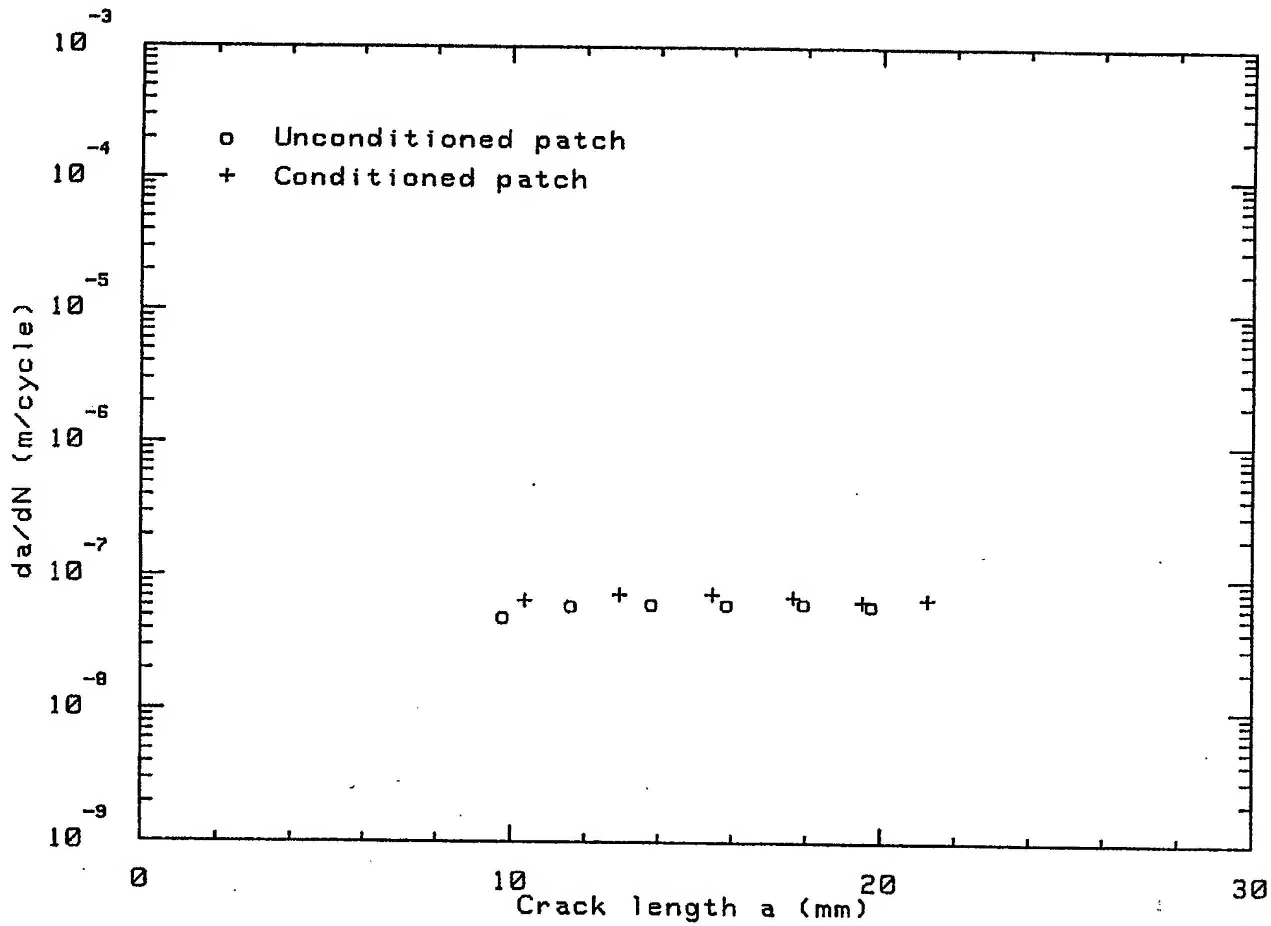
Effect of exposure to water at 74°C on rate of crack growth (double sheet specimen, BFRP patch with defects)



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Effect of reducing the adhesive shear modulus (G^a) for BFRP patch and a = 15mm

K_{min} and K_{max} [$MPa.m^{1/2}$]				
	no defects	10mm x 8mm	15mm x 12mm	20mm x 16mm
unpatched	1.3 - 26.7	1.3 - 26.7	1.3 - 26.7	1.3 - 26.7
patched & G^a = 0.4	10.1 - 16.9	11.0 - 18.3	12.9 - 21.5	13.8 - 23.0
patched & G^a = 0.6	9.4 - 15.7	10.2 -17.1	12.4 - 20.7	13.4 - 22.4
patched & G^a = 0.8	8.9 - 15.0	9.8 -16.3	12.1 - 20.2	13.2 - 21.9

ΔK [$MPa.m^{1/2}$] ($R = K_{min} / K_{max}$)				
	no defects	10mm x 8mm	15mm x 12mm	20mm x 16mm
unpatched	25.4 (0.05)	25.4(0.05)	25.4 (0.05)	25.4 (0.05)
patched & G^a = 0.4	6.8 (0.60)	7.4 (0.60)	8.6 (0.60)	9.2 (0.60)
patched & G^a = 0.6	6.3 (0.60)	6.9 (0.60)	8.3 (0.60)	8.9 (0.60)
patched & G^a = 0.8	6.1 (0.59)	6.6 (0.60)	8.1 (0.60)	8.8 (0.60)

$\Delta K^p / \Delta K^u$				
	no defects	10mm x 8mm	15mm x 12mm	20mm x 16mm
G^a = 0.4 GPa	0.27	0.29	0.34	0.36
G^a = 0.6 GPa	0.25	0.27	0.33	0.35
G^a = 0.8 GPa	0.24	0.26	0.32	0.35

DERA

8019707350/BL/POOLE

Effect of GRP layer (one ply Fibredux 924G) on patch repair

K_{min} and K_{max} [$MPa.m^{1/2}$]					
	a = 7.5mm	*a = 15mm*	*a = 30mm*	*a = 60mm*	*a = 90mm*
CFRP	10.6 -16.4	10.8 -16.9	10.3 - 16.7	8.3 - 15.6	9.2 - 25.7
CFRP + GRP	11.0 - 16.9	11.1 - 17.4	10.6 - 17.1	8.7 - 16.1	9.1 - 25.7

ΔK[$MPam^{1/2}$] ($R = K_{min} / K_{max}$)					
	a = 7.5mm	*a = 15mm*	*a = 30mm*	*a = 60mm*	*a = 90mm*
CFRP	5.8 (0.65)	6.1 (0.64)	6.4 (0.62)	7.3 (0.53)	16.5 (0.36)
CFRP + GRP	5.9 (0.65)	6.2 (0.64)	6.5 (0.62)	7.4 (0.54)	16.6 (0.35)

$\Delta K^p / \Delta K^u$					
	a = 7.5mm	*a = 15mm*	*a = 30mm*	*a = 60mm*	*a = 90mm*
CFRP	0.32	0.24	0.18	0.15	0.23
CFRP + GRP	0.33	0.25	0.19	0.15	0.23

Note: Assumed GRP properties: t = 0.253mm, E = 33GPa, $\alpha = 9.1 \times 10^{-6}$ (°C)-1

DERA

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Conclusions

- Patching increased fatigue life by a factor of at least 70
  - slightly longer lives were obtained for BFRP patches
  - bondline defects [8mm x 10mm] resulted in slightly shorter lives
- Disbonds grew from bondline defects adjacent to cracks, but not from those at tapered edges of patches
- Fatigue crack growth behaviour was predicted accurately by a 3-D FE/BE model
- The model predicted that rate of fatigue crack growth was
  - little affected by 8mm x 10mm bondline defects but significantly increased by 20mm x 16mm defects
  - little affected by a reduction in adhesive modulus from 0.8GPa to 0.4GPa
- Further work is required to study environmental effects

DERA

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Durability patch: high cycle fatigue damage dosimeter and vibration results of secondary structure repair

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Wes Owen¶ David Smith|| R. W. Gordon**

Abstract

The Durability Patch Program addresses the repair and life enhancement of secondary structure containing nuisance cracks which have been caused by resonant high cycle fatigue response to aerovibroacoustics. Design criteria include static strength, high cycle fatigue life, and crack growth rate. For this type of damage, safety of flight concerns are virtually non-existent, but maintenance and repair costs as well as recurrence rate can be high. The proposed repair methodology utilizes hot bonding in situ on the flight line. The Durability Patch (DPatch) consists of a bonded repair region which is an elastic elliptical laminate overlaid by and surrounded by a vibration damping treatment. In some configurations the transition from the elastic repair region to the damping region is accomplished by the use of a viscoelastic material instead of a structural adhesive in a portion of one layer; thus, the other layers are multifunctional. The bonded

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repair does not introduce stress concentrations, does reduce static and dynamic stresses, and does reduce crack tip stress intensities. The damping further reduces dynamic stresses and stress intensities. Damping is maximized within thickness and area constraints in order to enhance the life of adjoining structure with undetected damage. The life improvement goal is a factor of 600. Finite element analysis results comparing static and vibratory stresses will be presented. High cycle fatigue lives and crack growth rates will be compared. The design and use of a miniature autonomous Damage Dosimeter to obtain service temperature and vibration environmental data at low cost will be described. Dosimeter test results will be presented. Selection of structural materials and processes to attain a goal of field installation will be described. Comparison of analysis and laboratory results will be presented. DPatch configurations will be described and compared using a numerical measure of merit system. A repaired crack can continue to grow in service; the objective is to reduce the rate of growth by four decades.
Key words: Passive Damping, High-Cycle Fatigue, Bonded Repair, Cocuring, co-curing, viscoelastic material, composite material, finite element analysis, damping, modal strain energy.

1 INTRODUCTION

The Durability Patch Program addresses the restoration of structural integrity of cracked secondary structure induced by resonant high cycle fatigue. Secondary structure is any structure that does not involve safety of flight; examples are fairings, flap skins, etc. The program is based on adapting technology from three basic areas:

- bonded structural repair,
- vibration damping, and
- avionics.

These three areas each possess a large technology base and have achieved a threshold of maturity sufficient to support this program. A typical repair would be for a crack less than four inches long in 0.050-inch thick skin of the upper trailing edge of a wing. Nuisance cracking is a high maintenance and repair cost item. Typical sources of excitation are: pressure pulses from engine 1st stage compressor, jet engine exhaust, disturbed air flow behind

stores, separated flow on upper wing, air flow around open cavities, propeller tip vortices, etc. Typical locations of nuisance cracking are: flap skins, spoiler skins, rudder skins, aileron skins, weapon bay doors, wing trailing edges, etc. Of course there are other possible causes of cracking in secondary or lightly loaded structure besides resonant high cycle fatigue. [19]

High cycle fatigue life and crack growth rates are key disciplines in evaluating the longevity of structural repair. Methodology for calculation of resonant high cycle fatigue (HCF, sometimes called sonic fatigue or acoustic fatigue) life and associated crack growth rates used here is well established and consistent with standard industry practice [1, 2, 3, 4, 5]. It has been found that, in most cases, the HCF damage is due to linear resonant response in a single vibration mode; this implies that the vibratory stress is a narrow band random process. The threshold for number of cycles for high cycle fatigue is 10^6 cycles, which corresponds to 8.5 ksi (RMS) for 2024 aluminum alloy. Fatigue consists of crack initiation, propagation and final rupture. It is envisioned that because of the existence of a crack, the life is known and the baseline or unrepaired stress level may be calculated; one objective is to reduce the stress level such that life is enhanced. Stress levels will be reduced through beef-up and through vibration damping using viscoelastic materials (VEM). Strain and temperature as functions of time are available to the dosimeter and analysis is performed for the i-th frequency band, the j-th temperature band, and the k-th time increment. Time histories of temperature and the one third octave (or other) bands are recorded. Therefore, the cumulative damage may be calculated as a function of in service vibrational frequency and temperature.

Expert personnel visited four flight lines to assess available facilities, equipment, personnel skills, repair procedures, aircraft operations, etc. It was learned that almost all cracks are discovered and repaired before they reach a length of 4 inches. Also, scheduled flying and alert status constrains acceptable maintenance and repair techniques. A new repair technique would not be accepted if it required significantly more man-hours or clock time to implement. A typical small non-flush mechanically fastened sheet metal patch requires two man-hours to complete. This has been accepted as a goal for the present Durability Patch effort.

The total cost of repair of chronic nuisance cracking due to resonant vibration high cycle fatigue for all USAF aircraft is calculated to exceed 20 million dollars/year. The set of data is based entirely on professional judgment of the Non-Commissioned Officer In Charge (NCOIC) of sheet metal shops at selected flight lines. These cognizant expert personnel in all cases had

familiarity and experience of long standing.

The technology base for application of bonded repairs to aircraft structure has achieved a threshold of maturity sufficient to support this effort. Structural repair materials, structural adhesives, surface preparation techniques, design methods, and installation processing and procedures are well established. There are many applications performing satisfactorily in service, many of which are for primary structure. Bonded repair technology is well documented [6, 7, 8, 13]. One recommended design practice is that the patch match the extensional membrane stiffness of the baseline structure in order to avoid load attraction or shedding. Single sided repair results in eccentricity of load which induces bending stresses which must be accommodated.

Viscoelastic vibration damping technology has also achieved a level of maturity sufficient to support this effort [3]. Constrained layer damping is flying in service in air flow on external surfaces, some with an edge sealant and some with their perimeter adhesively bonded. The highest practical levels of damping will be used; this will enhance the life of the repaired skin, and will also enhance the life of adjacent bays of skin and substructure. This approach is judged to be appropriate in the context of demonstrated opportunity for improvement in durability with respect to nuisance cracking. Often the intrinsic damping is low; this fact makes the structure more susceptible to resonant high cycle fatigue cracking. This fact also increases the benefits of damping because RMS stress levels are highly dependent on modal damping. The dynamic magnification factor is inversely proportional to the modal damping. The modal strain energy (MSE) method has been established as the proper approach to calculate modal damping [9, 10, 11].

2 DOSIMETER

An adequate determination of damaging service conditions is required. Typically, a dosimeter would be flown on just one aircraft in a fleet for a case of fleetwide nuisance cracking. Depending on the location of another instance of cracking, dosimeter flights may or may not be required. Engineering judgment must be used to determine whether or not damaging services conditions are similar to known applications.

The Dosimeter has been conceived to gather service environmental data with regard to suspected resonant HCF cracks as economically as practical. Dosimeter requirements are that service data be gathered, processed

and stored to permit: the design of a damping treatment (which requires the knowledge of the vibration frequency and temperature at which damage is being accumulated in service), a valid quantitative comparison of structural life before and after Durability Patch (DPatch) installation, and any convenient additional diagnostic information.

The Dosimeter is a key component of the process to design and install the most effective patch possible. In order to provide this function the dosimeter must meet several goals:

1. The dosimeter should be simple to install/remove on the widest practical variety of aircraft and locations.
2. The dosimeter should autonomously measure high frequency strain and temperature while the aircraft is operational.
3. The dosimeter should be affordable.

In order to build a useful dosimeter, these goals must be merged with practical cost and operating procedure considerations. As with many other "engineered" products, the finished dosimeter must delicately balance conflicting requirements in order to achieve the best possible performance across the widest variety of aircraft and applications.

In order to achieve these lofty goals a design that is both modular in components and logic has been adopted. Figure 1 diagrams the major logical components in the dosimeter. The first component, the sensors, are comprised of up to three axial strain gages, and a single temperature gage. Both the strain and temperature gages are inexpensive and relatively simple to install. Up to 50 feet of lead wire is allowed between the sensors and the dosimeter processor and power-supply, without adjustment to the sensor gage-factors. This design has several advantages: the dosimeter can be installed by almost any mechanic with strain gage experience, the processor and battery units do *not* have to be micro-miniaturized, and improved signal-to-noise ratio over conventional strain gage signal conditioning methods [18].

The processor and data-storage unit is powered by its own battery for completely stand-alone operation. This has many advantages, and one disadvantage. For advantages there are: the ability to operate on any aircraft, the ability to tailor the battery for different packaging or power requirements, and the ability to install the battery in a location separate (and maybe distant) from the processor and data storage unit. The one disadvantage:

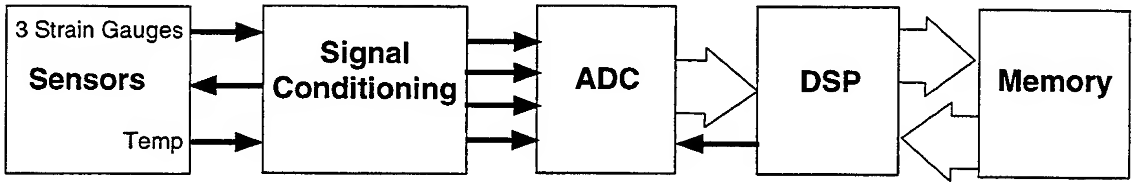

Figure 1: Damage Dosimeter Data/Logic Flow

batteries low tolerance to cold temperatures, is somewhat mitigated by the ability to install the battery distant (warmer) from the processor unit, and the existence of lithium manganese dioxide batteries. Commercially available (Duracell) $LiMnO_2$ batteries have been tested down to -40^o Celsius with excellent results. Although these batteries are not rechargeable, they are in-expensive.

The processor and data storage unit (PDSU) provides several services: autonomous operation, analog-to-digital conversion of the sensor signals, computational operations on the time-series data, intelligent storage of pertinent data, and communications with a laptop computer for data download, and program upload. The component central to the PDSU is the Analog Device ADSP-2181 Digital Signal Processor (DSP) chip. This DSP chip offers fast integer and fixed-point computations, along with low power consumption. The PDSU contains 4 mega-bytes of flash e-prom for data and program storage. This memory is non-volatile in order to minimize the potential for data loss in the event of low-power or unforeseen circumstances. The PDSU is designed to operate for approximately 12 hours continuously, and up to 2 weeks in mixed operational and standby modes.

Figure 2 shows a Damage Dosimeter in its nominal package. The enclosure is roughly 5.75in by 4.25in by 1.25in, weighing roughly 1.25lbm. There are two connectors. One for the gages and battery, the other for serial communications with a computer. The nominal battery package will be approximately 4.5in by 3.5in by 2.1in, weighing approximately 1.5lbm.

Once the dosimeter has been installed on an aircraft, and connected to its battery, it begins its duty cycle. This duty cycle consists of waking itself every N (configurable) seconds, and acquiring one "block" of data. The root-mean-square (RMS) strain levels are computed from the time-series data, if the RMS strain levels are high-enough, then the dosimeter continues to operate until 10 seconds, or blocks, of "quiet" data are recorded. At this point, the dosimeter sleeps for N seconds, and then starts the duty cycle all over again. This process is repeated until the power is removed from the dosimeter. Since

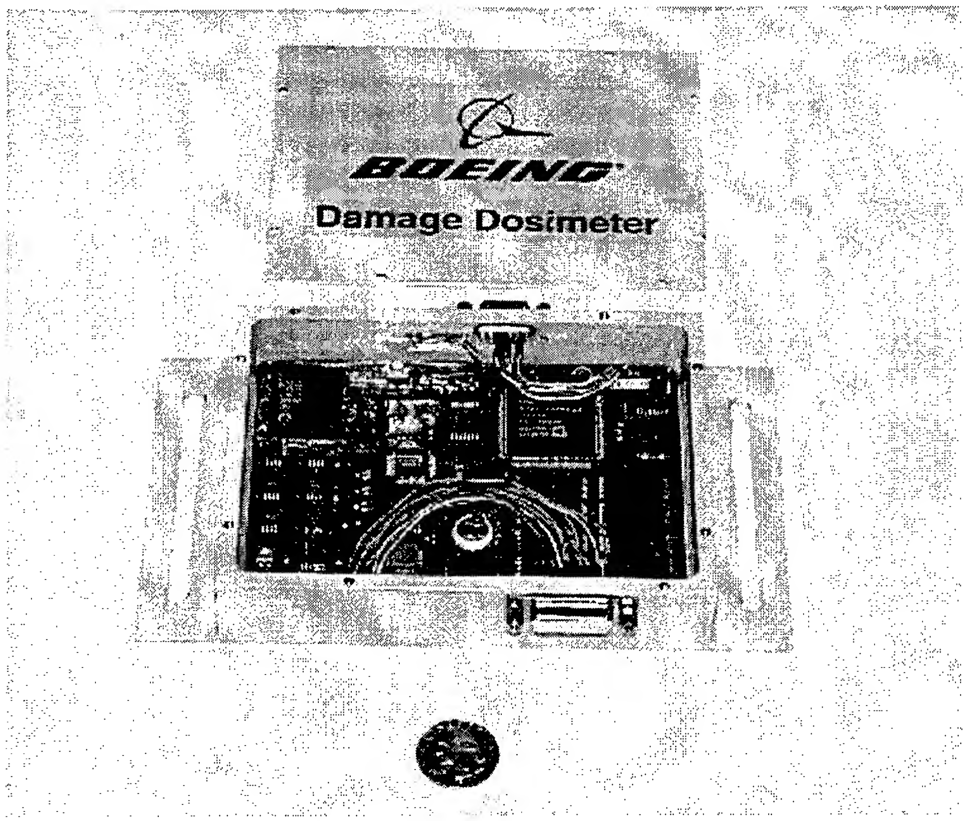

Figure 2: Damage Dosimeter Electronics

the dosimeter's memory is non-volatile, data will not be lost when power is removed.

The dosimeter records a time-history of strain each second, and processes the time-history for the remainder of each second. This method is valid as long as peak value strain detection is not important, which is the case with HCF cracking problems. Typically the structure is responding in a steady-state fashion. From each time-history, the RMS strain in certain 1/3 octave bands is computed, along with minimum and maximum strain values and temperature. These data are saved in memory, along with typical and worst-case sample strain time-histories. The maximum overall level is not expected to exceed 3000 micro strain peak. The frequency range of interest is from 44.7 Hz (the lower limit of the 50 Hz band) to 2239Hz (the upper limit of the 2000 Hz band).

3 CONFIGURATIONS

The following design philosophy is summarized by these factors:

- Restore static capability
- Enhance life
- Minimum quality assurance/inspection
- Cost savings
- Ease of installation
- Aerodynamically smooth

The static capability of the structure must be restored. It is well known in bonded repair that the extensional membrane stiffness of the original skin should be closely matched to avoid load attraction and load shedding. Of course this is true only if the structure carries significant stress. Regardless, the repaired structure must be capable of carrying any applied loads. Since the existence of nuisance cracking demonstrates the opportunity for improvement in durability, the local flexural stiffness should be enhanced in order to better withstand loads.

The life of a properly designed and installed bonded repair will exceed the life of the undamaged baseline structure, although in this case that is known to offer opportunities for improvement. The DPatch must withstand moisture for decades, must reduce stress intensity and consequent crack growth rate, should reduce static stresses, and must reduce dynamic stresses. These points suggest no stress concentrations or hard points, vibration damping, and high tolerance of large disbonds/porosity.

The context may be summarized by the following list of points:

- Need for restoration of structural integrity of cracked secondary structure
- Demonstrated opportunity for improvement in durability
- Bonding installation on flight line by inexperienced personnel
- Minor direct consequences of large disbonds
- Opportunity for developing bonding personnel/service experience

There are many advantages to a repair configuration which enhances flexural stiffness: reduced eccentricity of the load path due to in-plane loading;

reduced patch bending; reduced bending of the original skin; reduced peel stresses; reduced stress intensity at the crack tip. Increased flexural stiffness reduces the curvature of the original skin at the crack due to vibration, reduces stresses in the skin, patch, and adhesive, and reduces stress intensity factor for vibration. The feature of primary importance here is flexural stiffness; if the modulus of the repair material is such that there is no thickness available for a sandwich core, the flexural stiffness is at a practical maximum.

In one instance a bonded repair patch of thin boron fiber was installed on the external surface of lower nacelle skin; however, the cracks continued to grow [15]. This situation emphasizes the need to consider HCF and crack growth rates of repaired structure.

Several configurations have been defined for a preliminary evaluation of their static and vibratory structural characteristics. In all configurations, the elastic repair region is elliptical. It is adhesively bonded to the original skin centered on the crack location. The elliptical elastic repair region is surrounded at its perimeter by damping, and it may or may not have a damping overlay.

One candidate configuration has been identified; the surface preparation is grit/blast silane, then primer. The film adhesive is EA9696. The room temperature viscoelastic damping material layer is Soundcoat Dyad 601. Then layers of fiberglass epoxy prepreg are installed. Cure is with vacuum bag and heat blanket at 250^o F. This requires approximately 8 hours elapsed time. For most locations, the use of a heat blanket requires that the aircraft be defueled and purged. If this process is performed properly, it is believed to have longevity exceeding that of the airplane.

Other candidate installation procedures and configurations are under consideration at this time. The goal of these other configurations is to reduce the total installation time to something that is competitive with metal repairs. Furthermore, for maximum cost benefit (savings) the configuration must be usable on the flight line by personnel with minimal training. New surface preparations currently under development, such as Sol-Gel [20] have demonstrated the potential for dramatic reductions in surface preparation time, personnel training, and equipment. The challenge is to take advantage of the low in-plane stresses present in most secondary structures, *without* compromising repair durability and longevity.

There are some remaining issues. One is the maximum permissible thickness for the Durability Patch; among some circles, it is believed that 0.125 inch thickness is acceptable, especially away from leading edges, because the boundary layer is much thicker than that. Computational fluid dynamics

studies have been performed on other types of situations which so indicate. Another issue is how a vacuum can be maintained over a row of fasteners and a crack. Repaired crack growth rates for in-plane loads and out of plane vibration are unknown, although they are less for bonded repair than for conventional mechanically fastened patches. Survival of Patches at high Mach, high temperatures and the regaining of the damping, strength, and stiffness functions at service damaging conditions are required.

A preliminary finite element analysis (FEA) was performed. The baseline model was a bare clamped-clamped beam 10.5 inches long of unit width. This is representative of a unit width strip of the test article described in Section 5. For all configurations analyzed, the stresses in the original skin are very low and as a consequence have a Life Improvement Factor (LIF) of significantly greater than 600. (A LIF of 600 results if the HCF crack is detected at 100 hours, and there is a life of 60,000 hrs required). If there is a skin panel on the other side of the substructure from the crack it will be covered by a portion of the DPatch; it probably has accumulated HCF damage before the DPatch is installed, although it is probably not detectable. It is anticipated that no crack will ever occur in that panel. Any skin panels adjacent to these two will be damped through coupling and will also have their life extended significantly. Also in all configurations, the stress level in the adhesive is well within the linear elastic range; as a consequence, the HCF life of adhesive will be greater than the remaining life of the aircraft. Only fiberglass configurations are being considered.

Single sided repair leads to load path eccentricity which gives rise to secondary bending. Max stresses in the original skin are approximately twice the remote uniaxial stress. For the present purposes of repair and life enhancement of lightly loaded structure, the in-plane loading will be taken as zero. This is a valid approach because the structures of interest are fairings and similar lightly loaded structure where the in-plane loading is very small.

Strength and stiffness consideration leads to the requirement that the tension ultimate divided by the repair thickness must be greater than or equal to that of the baseline; this is satisfied when any aluminum is repaired by any woven fiberglass. In unsupported single lap joints the overlap should be 80 times the thickness of the original skin per Hart-Smith as reported in Baker Jones [6] page 32. When it is considered that in-plane loads for the secondary structure of interest here are a small fraction of ultimate structural capability, the design for static strength is intrinsically satisfied.

It is anticipated that a layer of structural film adhesive will be in contact with the original skin. Next a layer of VEM with the elliptical hole will be put in place. This arrangement provides a maximum of moisture protection of the aluminum surface to adhesive bond. It is also anticipated that a local thickening of adhesive will occur as recommended by Hart-Smith and reported in Baker Jones [6] page 32 at the perimeter of the elliptical repair region.

4 MATERIALS/PROCESSES

A goal is that the installation of the bonded repair will be on the flight line at an operational base; this is considered to be somewhere between very challenging and unrealistic/impossible by many experts in bonded repair of primary structure. It is noted that the direct economic and technical consequences of extensive disbonding of a Durability Patch is minor and that this type of repair is a very low profile application. This situation may be used to good advantage in order to maximize benefits.

Very importantly, the DPatch must offer an attractive option (relative to conventional techniques) to the potential user, or it will not be accepted. This means that it must be simple to install, require no more man-hours than conventional repair, require no more clock time, no more requirements for aircraft environment, environmentally safe, etc. It must result in net cost savings with no adverse effects.

In order to minimize costs, there should be a minimum of quality assurance and in-service inspection. Measurement and recording of temperatures during cure, and a visual and coin tap afterward are probably the only requirements. No scheduled in service inspection is being considered.

One longevity aspect not yet covered is that of the structural adhesive. Surface prep and adhesive combinations subjected to moisture over decades in service are key to the longevity of the DPatch. Regardless of any moisture barriers, eventually moisture will intrude into the entire bond line. At this juncture, the best indication of longevity of the bond line subjected to moisture is the wedge test; criteria are both a threshold of crack growth in the wedge as well as a cohesive failure mode. Bonding longevity is indicated by crack growth of the wedge test and also by the necessity of a cohesive failure mode.

On the flight line, the 1750 sealant is currently used much of the time when performing mechanically fastened patches. It is a two part compound and is very messy to use.

The film adhesives being considered are Cytec FM 73, 3M AF 163-2, and Dexter EA 9696, all nominally 250^o F cure. It is believed that all of these may be cured at a somewhat lower temperature with a minimum sacrifice in mechanical properties and environmental longevity. It is desired to use pre-preg resin consistent with these characteristics.

Achieving 250^o F on an airplane for cure of the bonded patch to aluminum skin/substructure may be difficult, whereas 200^o F is fairly easy. The planned cure temperature will routinely be 250^o F except when there is a concern about moisture already trapped in the structure; in this case we will cure at 200^o F for a longer duration. The expectation is that any and all 250^o F amine (not anhyride) cure epoxy resin for fiberglass pre-preg meeting Boeing BMS 8-79 will be compatible with FM73, AF163, and EA9696.

5 TESTING

An electrical chassis box 8 x 17 inches of 0.050 aluminum was selected as a test article; a substructural element was bonded to the underside to provide two skin bays.

Ten viscoelastic damping materials were selected as candidates for use in the Durablity Patch. The complex modulus of each was determined in both the non-co-cured and co-cured[1] configuration. The non-co-cured was as supplied and then mounted in the test configuration. The co-cured configuration was fabricated into a laminate with a layer of fiberglass prepreg, a layer of VEM, a layer of structural adhesive, and another layer of fiberglass and then subjected to a 250^o F cure cycle; note that the VEM had surface to surface contact with structural adhesive on one side and with fiberglass prepreg on the other. In some cases the polymers interact during the cure cycle and change the dynamic mechanical properties. Rectangular specimens were cut from these cured laminates and mounted in the test configuration. A typical set of complex modulus data is shown in the reduced frequency nomogram format useful for design (Figure 3).

[1] Co-cured means that the adhesive is cured in surface-to-surface (not edge-to-edge) contact with the VEM.

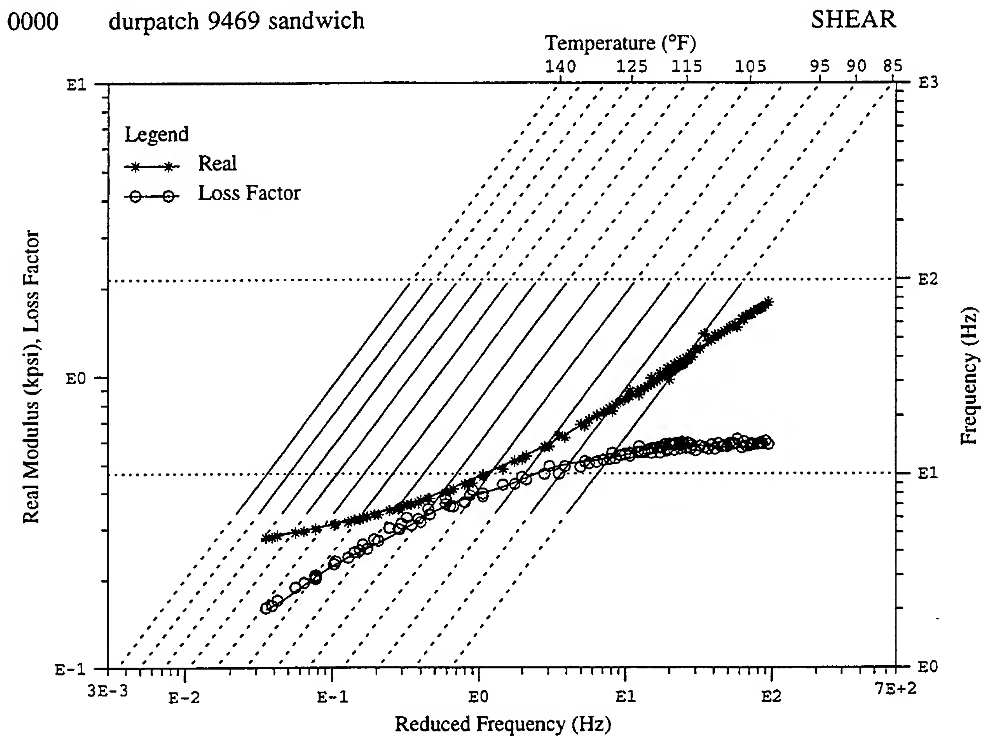

Figure 3: Co-Cured Laminate with 3M9469 VEM

To obtain the fiberglass modulus, a laminate consisting of ten layers of fiberglass prepreg was cured. The measured thickness was 0.125 inches; a beam 1 x 12 inches was cut from the laminate and the frequencies of the first two vibration modes was measured. From these frequencies, a modulus of 2.6 msi (million pounds per square inch) was determined.

The vibration test articles were 8"x17" aluminum boxes with a 3" skirt to which a "T" was bonded to form two bays, one of which is 8"x7.5". The configuration of the Durability Patch both analyzed and tested consisted of: a) a layer of structural adhesive; b) a layer of VEM with an elliptical cut-out centered on the crack location; c) 10 layers of fiberglass prepreg. A finite element analysis of the chassis-T was performed which obtained the frequency and the modal strain energy (MSE) of two lowest frequency modes as a function of the shear modulus of the viscoelastic material (VEM) layer. The results are presented in Table 1 and in Figure 4. The frequency and MSE as a function of shear modulus are independent of the particular VEM under consideration. Figure 5 presents measured frequency response functions for

the chassis-T with a particular VEM; from this information, modal damping and frequency were determined as a function of temperature and plotted in Figure 6. The results of Figures 3, 4 may be combined to compare predictions with experiment in Figure 6. A value of shear modulus from Figure 4 corresponds to a unique value of reduced frequency and material loss factor in Figure 3. The modal damping is calculated from MSE and material loss factor. The temperature is calculated from frequency and reduced frequency. Figure 6 includes analysis curves for the same VEM in a non-cocured condition. Note that in general there is a change in peak damping and in the corresponding temperature. Table 2 summarizes the information. The correlation between analysis and measurement in Figure 6 is excellent. One purpose of the Durability Patch Program is to develop and offer an attractive alternative to the conventional repair of cracked secondary sheet metal structure. For this to happen, it seems that the procedure must be performed on the flight line in shortly after discovery of any cracks, and that it should require less person-hours and clock elapsed time.

VEM	Mode 1				Mode 2			
G	Frequency	MSE	modal damping	temperature	Frequency	MSE	modal damping	temperature
psi	Hz	%	%	°F	Hz	%	%	°F
40	200.6	16.4	--	--	246.8	12.9	--	--
100	218.8	21.0	--	--	265.3	18.2	--	--
200	235.6	21.3	--	--	283.5	19.6	--	--
300	245.8	20.0	4.0	180	294.9	18.9	3.8	182
500	257.8	17.2	6.9	135	308.7	16.8	6.7	138
800	267.5	14.2	7.8	112	320.1	14.0	7.7	115
1000	271.5	12.8	7.7	106	324.9	12.6	7.6	108
1500	278.0	10.3	6.2	94	332.5	10.2	6.1	97

Table 1: Modal Damping and Temperature

6 DISCUSSION

A major benefit of the DPatch is the minimal potential for additional damage because repairs are made in-situ which minimizes handling damage. The Durability Patch Program has additional payoff beyond the program and repair in that service experience for bonded repair and a pool of personnel skills will be developed. Furthermore, experience is provided for future applications of micro data collectors, analyzers, loggers, eg, health monitoring.

VEM	temperature of peak damping		peak damping ratio
	not cocured	cocured	not cocured/cocured
3M 9469	74°F	120°F	2.3
3M 9473	79°F	103°F	1.9
3M 468	85°F	102°F	1.6
Dyad601	62°F	62°F	1.0
Dyad606	132°F	132°F	1.0
Dyad609	148°F	148°F	1.0
Avery3099	98°F	96°F	1.0
Avery1125	TBD	62°F	TBD
Avery1191P3	TBD	TBD	TBD
3M 9245	TBD	92°F	TBD
Densil2078	TBD	172°F	TBD

Table 2: Temperature of Peak Damping for Candidate VEMs

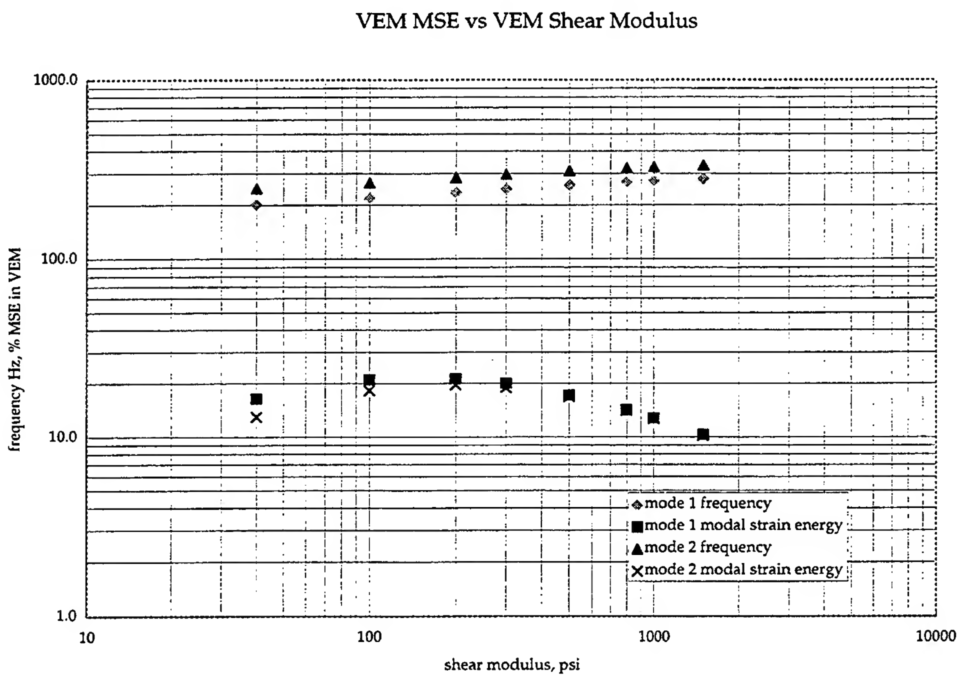

Figure 4: Modal Strain Energy and Frequency versus VEM Modulus

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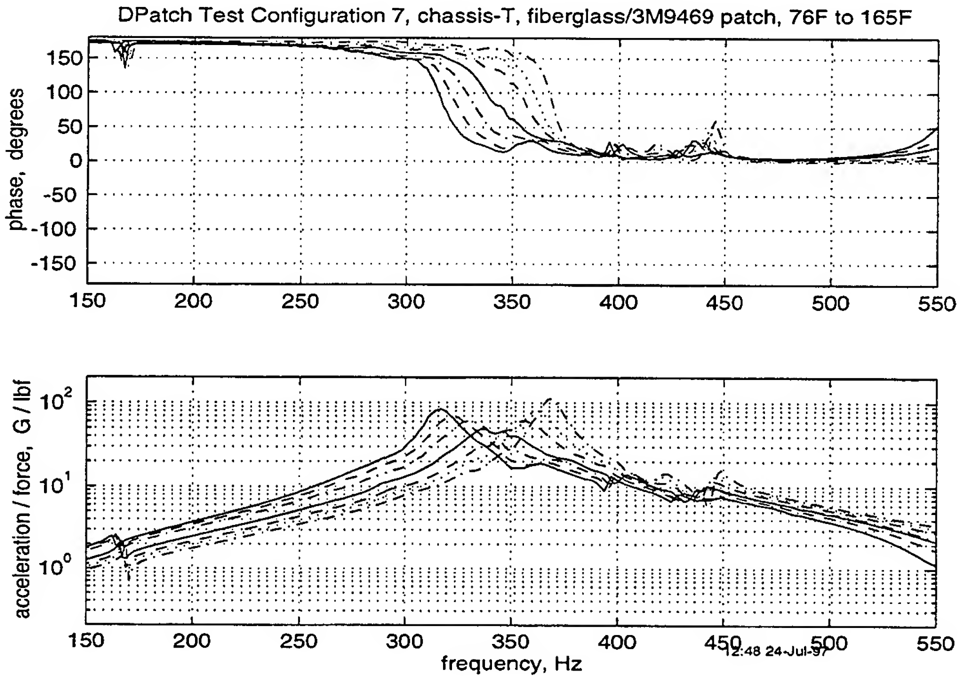

Figure 5: Measured FRFs for Temperatures Between 76° and 165° F

6915

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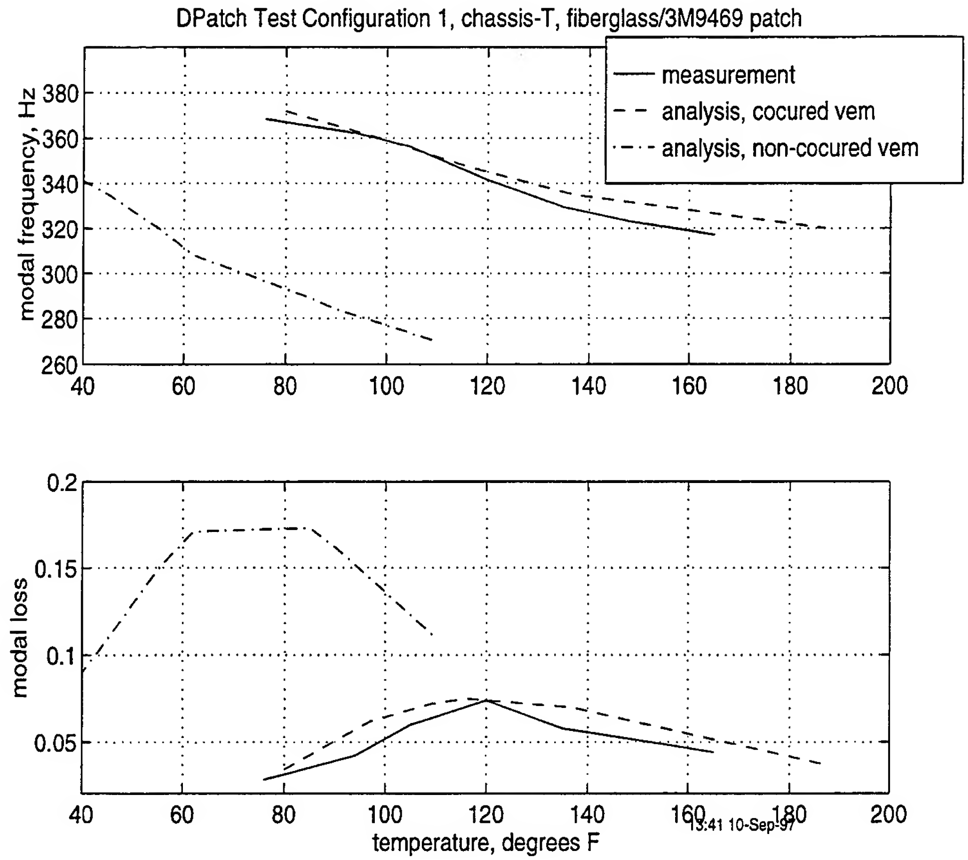

Figure 6: DPatch on Chassis-T Test Article, Measured and Predicted

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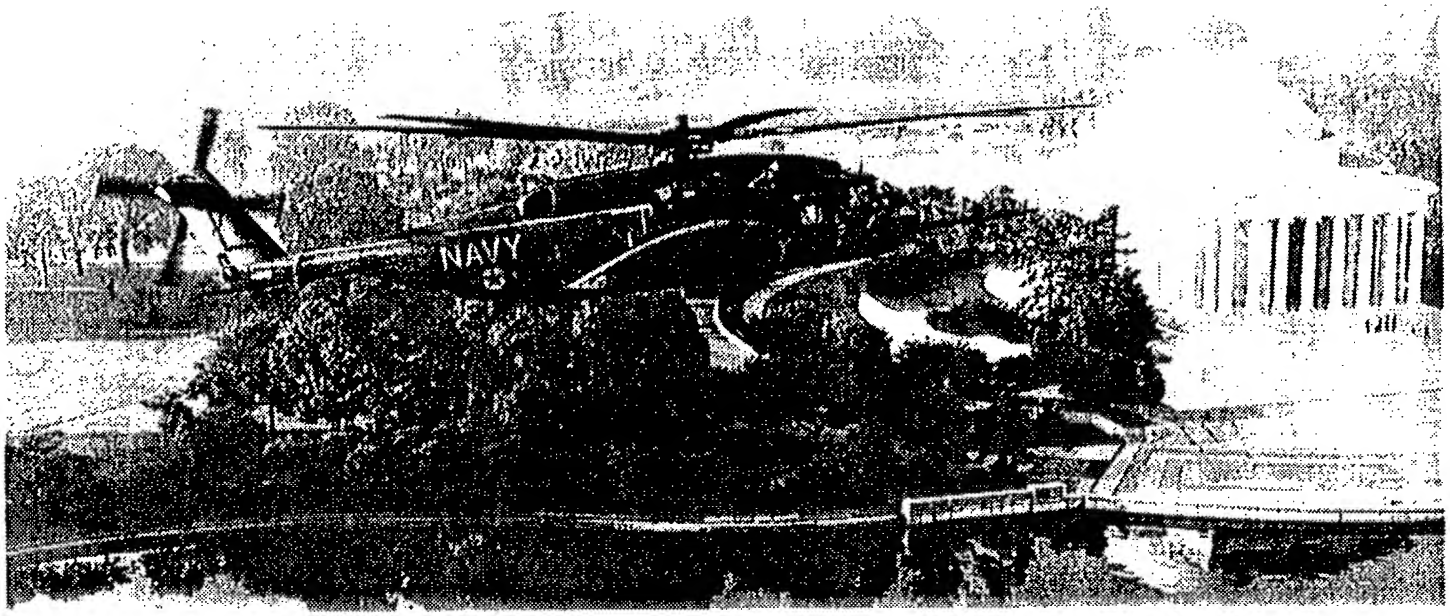
Acknowledgment: Support of the US Air Force is gratefully acknowledged.



MH-53E COMPOSITE SPONSON DAMAGE TOLERANCE AND REPAIR PROGRAM

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Cherry Pt, NC

Alan Dobyns
Sikorsky Aircraft
Stratford, CT

MH-53E COMPOSITE SPONSON DAMAGE TOLERANCE AND REPAIR PROGRAM

Program Objectives:

- To define and validate damage tolerance and repairs for the MH-53E composite sponson.

- Provide the Navy with the analytical methods, test results, and repair methods to write a repair work package for the sponson.

- Provide the Navy CFA Engineering with the information needed to analytically design unique repairs for the MH-53E sponson.

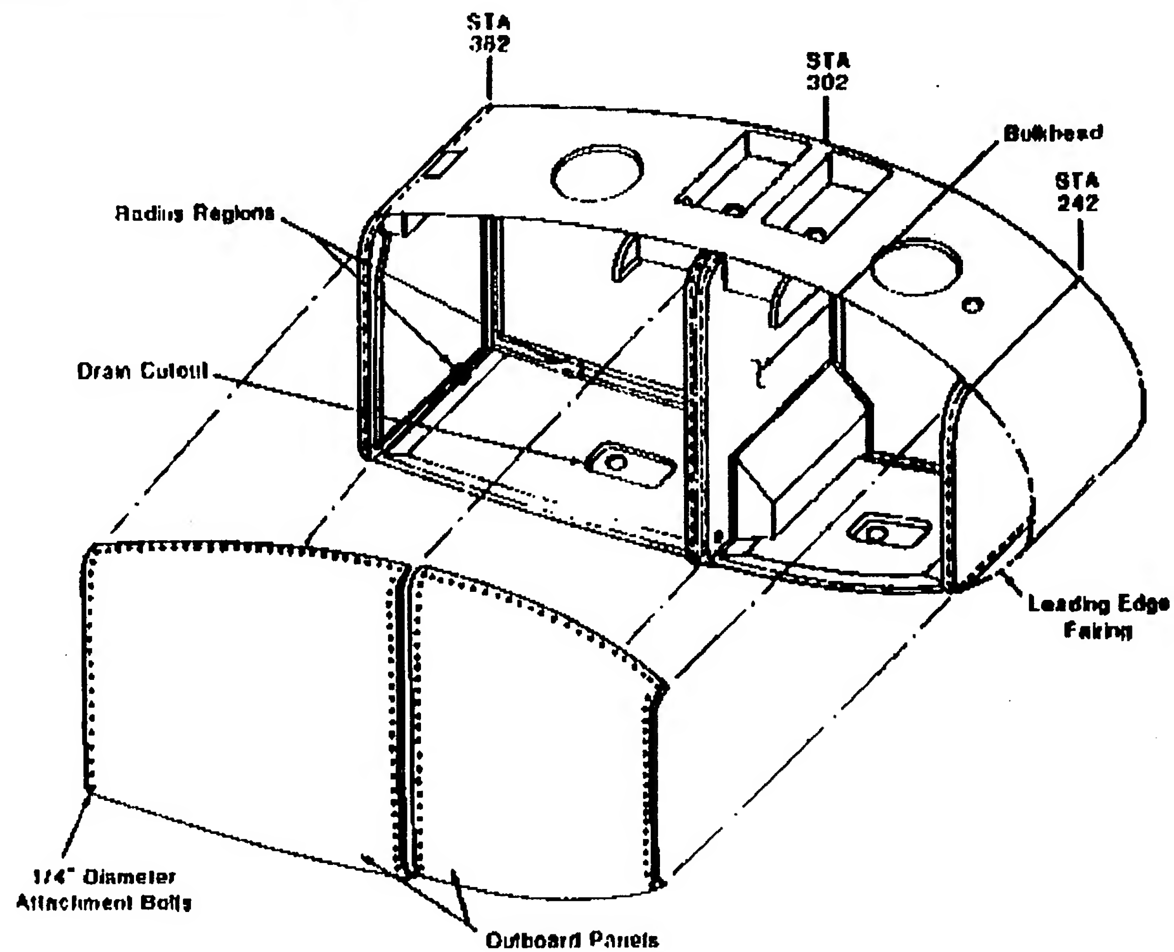
MH-53E COMPOSITE SPONSON DAMAGE TOLERANCE AND REPAIR PROGRAM

Task Breakdown:

- Strain Maps and Zone Breakdown
- Damage Tolerance Analysis
- Repair Methods and Analyses Techniques
- Repair Design and Analysis
- Test Matrix and Plan
- Test Specimen Development and Validation
- Specimens Fabrication and Repairs
- Test Program
- Analysis and Test Comparison
- Final Report

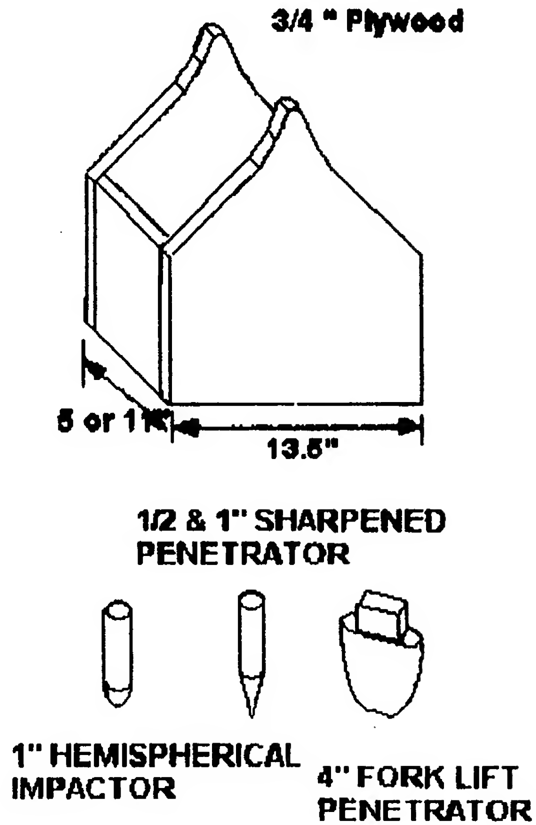
MH-53E COMPOSITE SPONSON DAMAGE TOLERANCE AND REPAIR PROGRAM

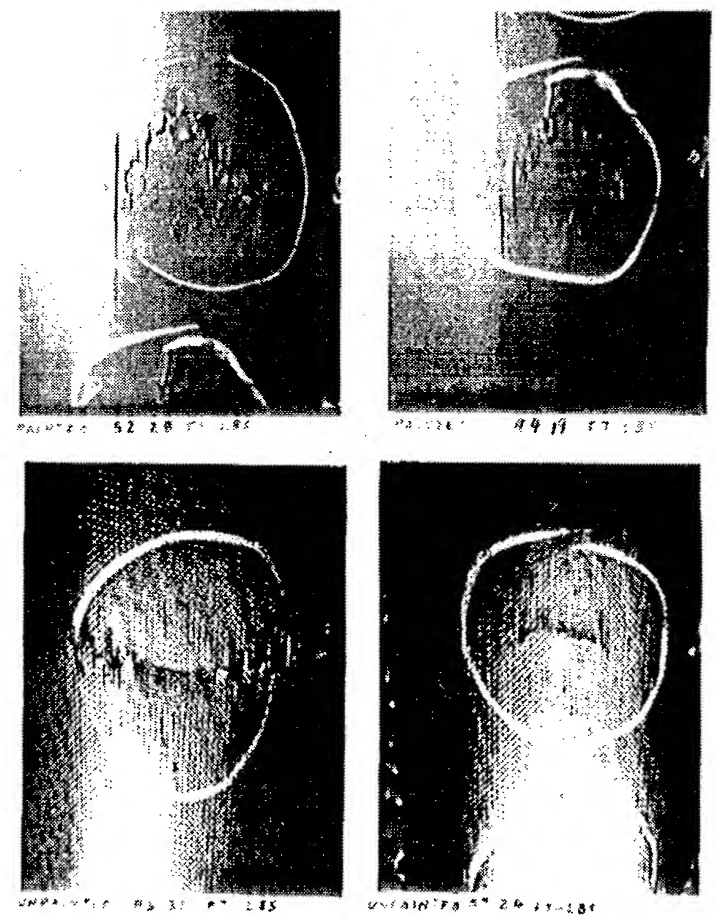
Zone Breakdown:



MH-53E COMPOSITE SPONSON DAMAGE TOLERANCE

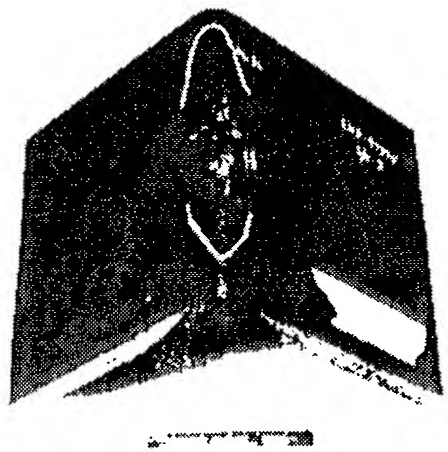
- THRESHOLD OF VISIBILITY

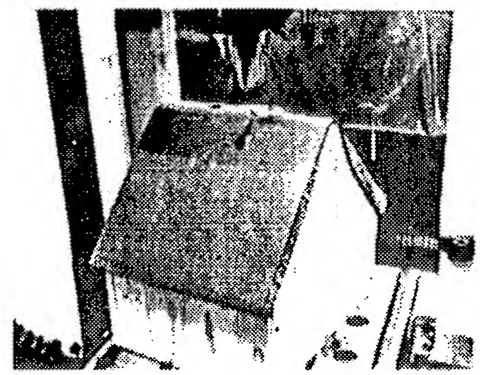


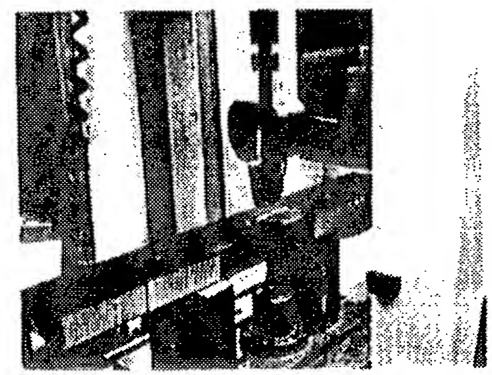


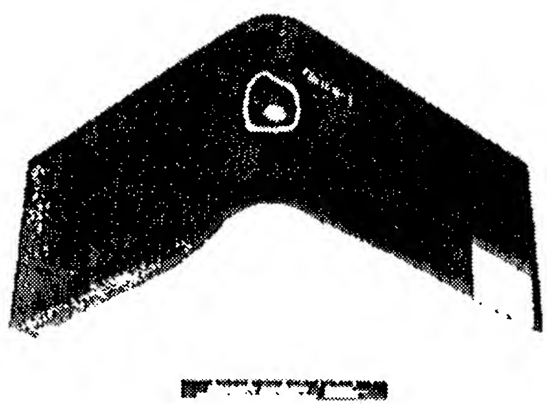
MH-53E COMPOSITE SPONSON DAMAGE TOLERANCE

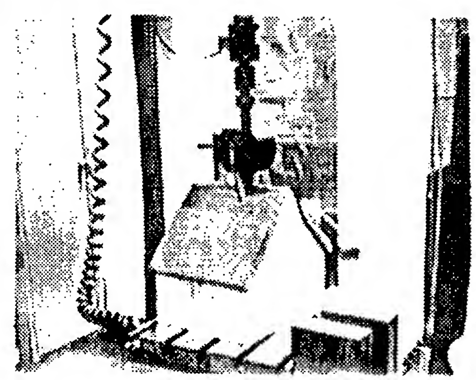
- UNREPAIRED DAMAGE

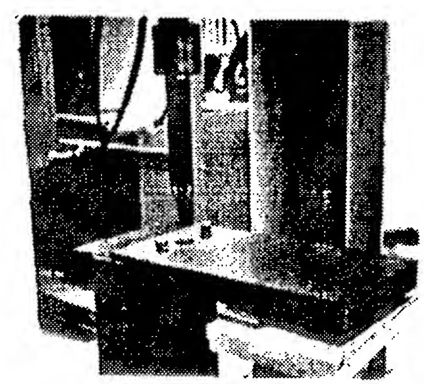












MH-53E COMPOSITE SPONSON REPAIRS

REPAIR METHODS USED

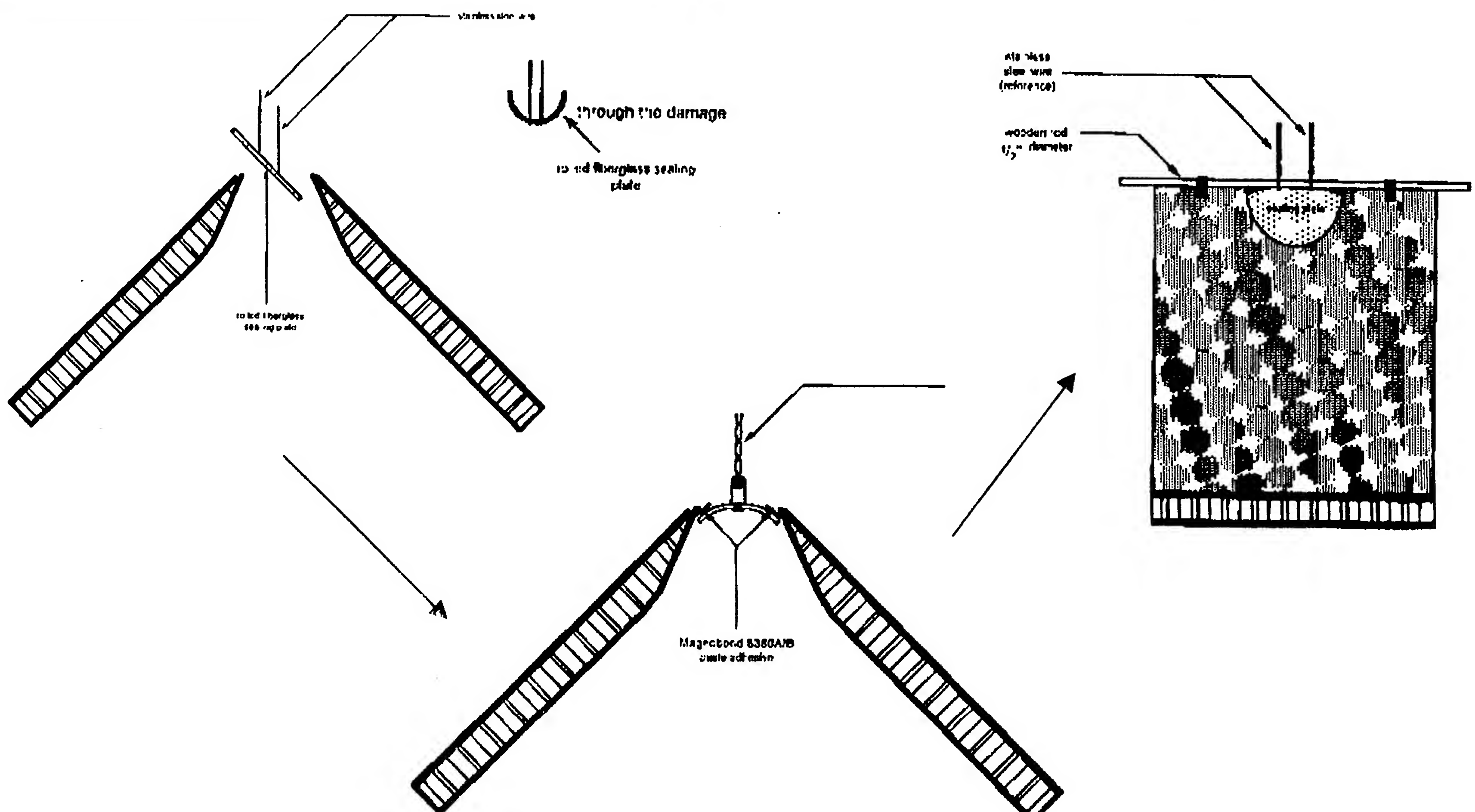
- 101SPECIMENS WERE REPAIRED FOR THIS PROGRAM
  - The majority of repairs were performed by MH-53E fleet personnel at Sikorsky Aircraft.
- Repairs were done to specimens representing each sponson zone.
- Repair types included:
  - One Sided Access - Backing Plates
  - Two sided Access
  - Wet Layups - Double Vacuum Debulk
  - Precured Patches
  - Elevated Temperature Cure Repairs
  - Room Temperature Cure Repairs
  - Bonded Metal Repairs
  - Bolted Repairs

MH-53E COMPOSITE SPONSON REPAIRS
REPAIR MATERIALS USED

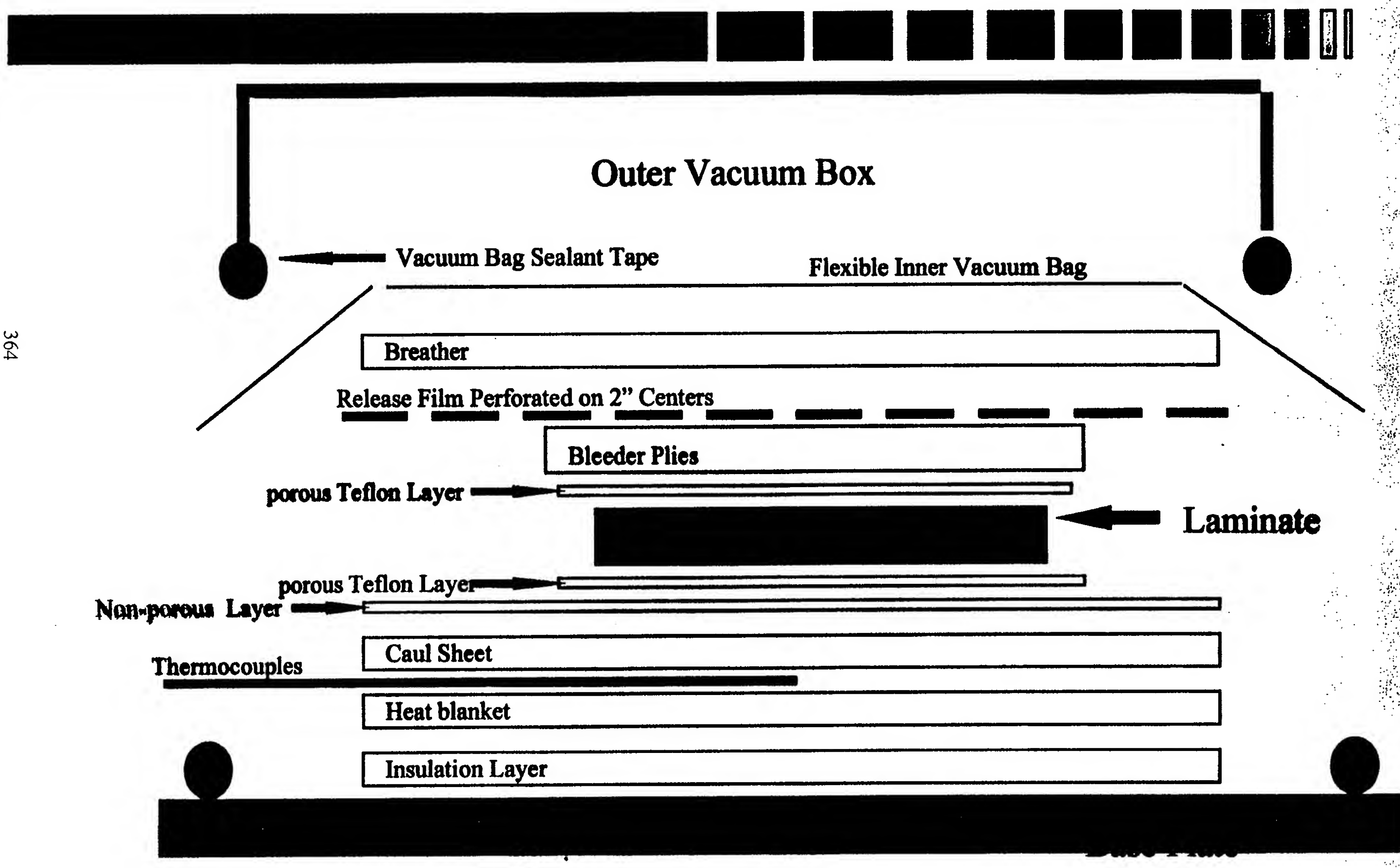
- Primary Resins and Adhesives:
  - Magnobond 6363 adhesive was used to bond patches.
  - Hysol EA 9390 was the primary laminate resin.
  - Room temperature cure repairs use Magnobond 6380 adhesive and Hysol EA9396 resin.
  - Hysol EA9396.6MD was used to bond honeycomb core.
- Fabric:
  - AS4 plain weave carbon composite was used for wet lay-up repairs.
- Prepreg:
  - C3K/5225 precured 2 ply sheets were used for precured patches.

MH-53E COMPOSITE SPONSON REPAIRS

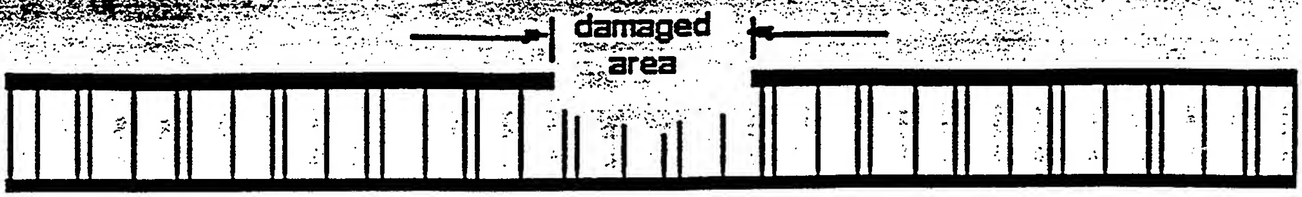
ONE-SIDED ACCESS

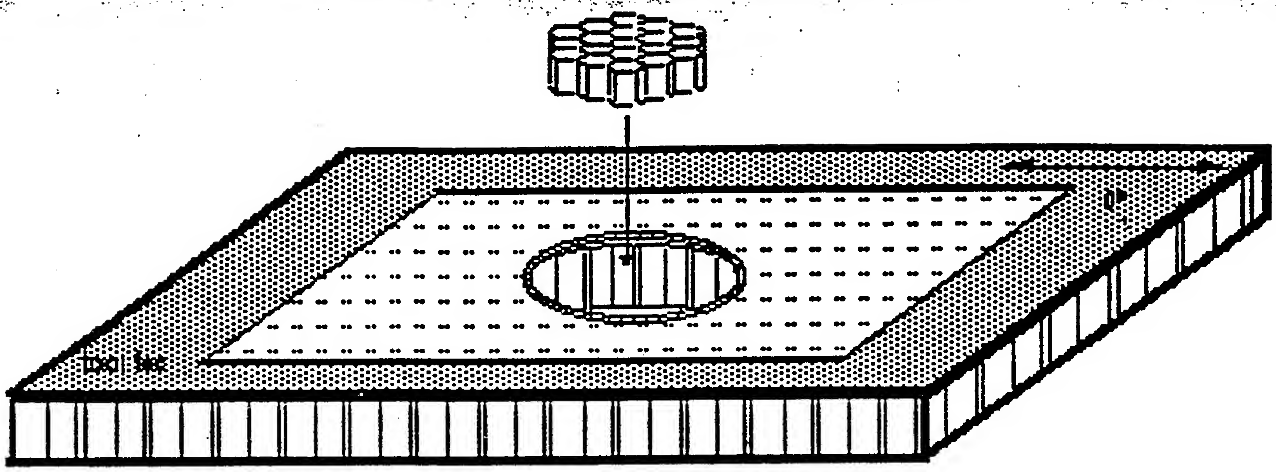


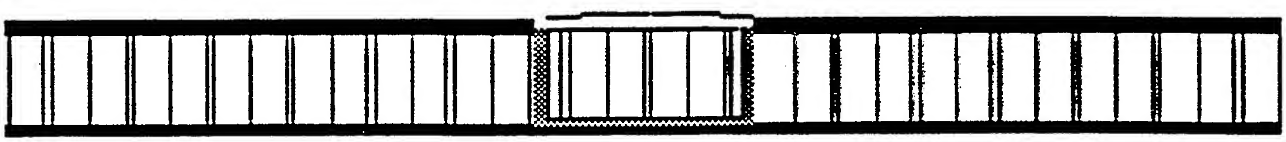
MH-53E COMPOSITE SPONSON REPAIRS
DOUBLE VACUUM DEBULK SET-UP



HONEYCOMB SANDWICH PANEL REPAIR PROCEDURES







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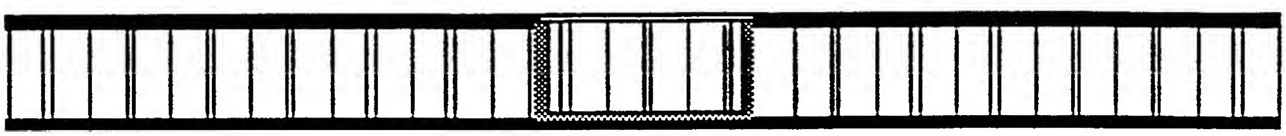
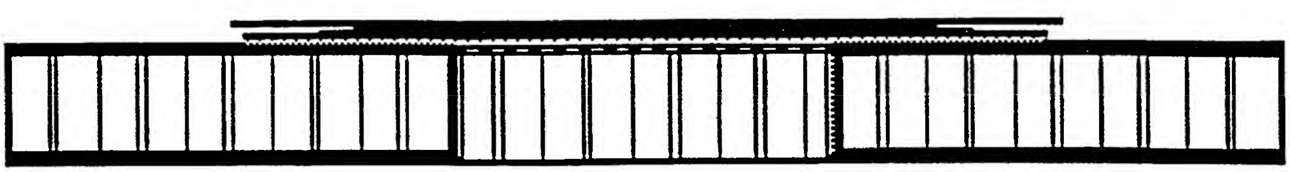
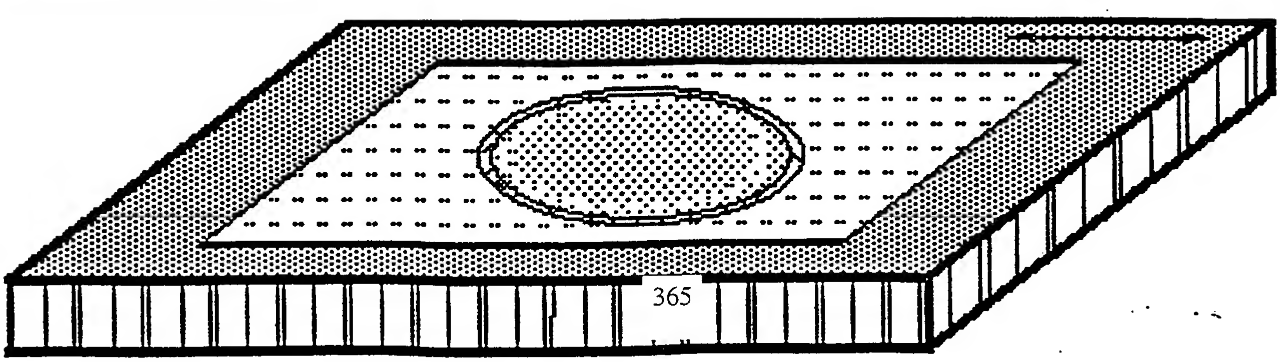
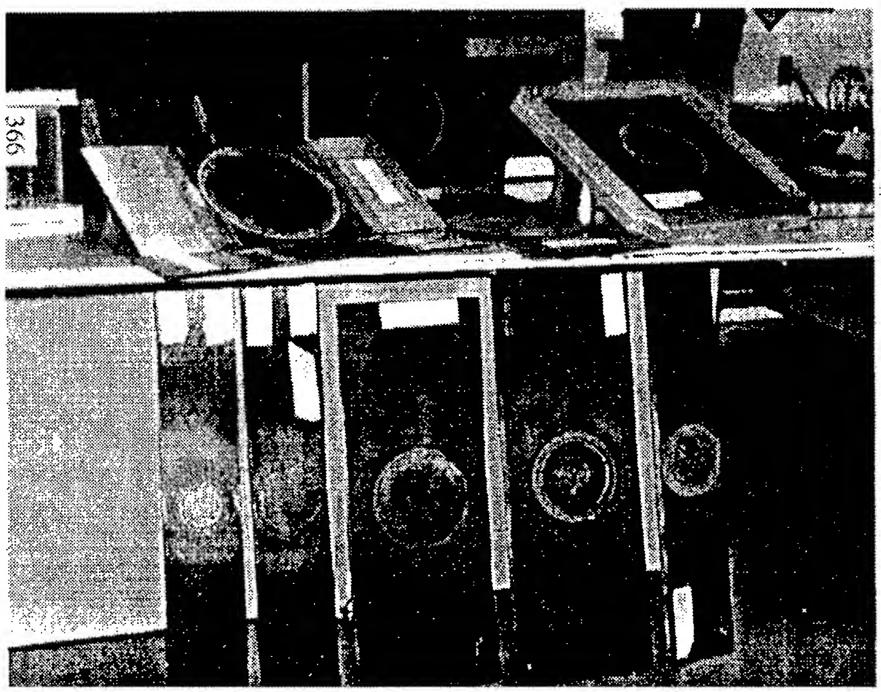


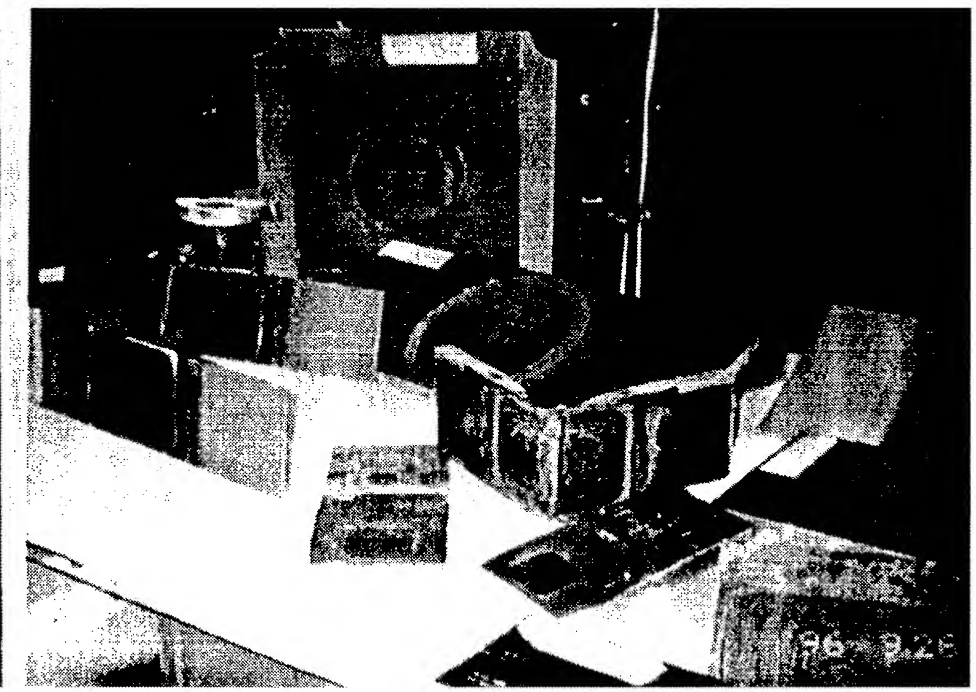


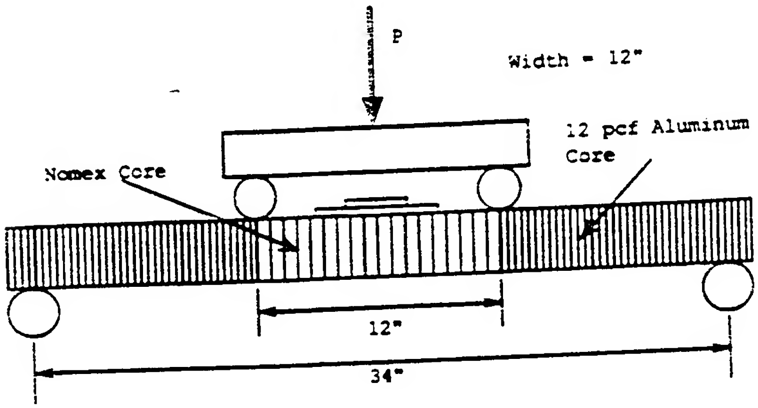


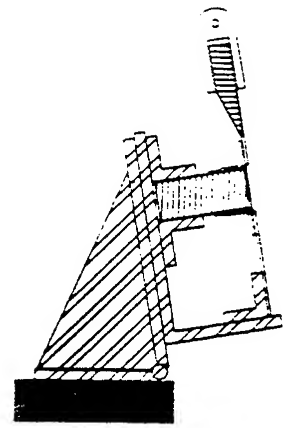
MH-53E COMPOSITE SPONSON
REPAIRED SPECIMENS

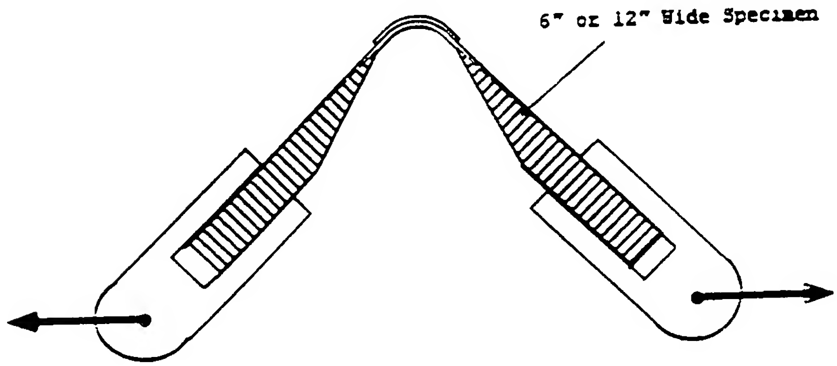
- **REPAIRED SPECIMEN**

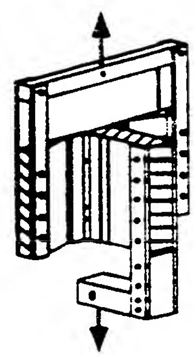


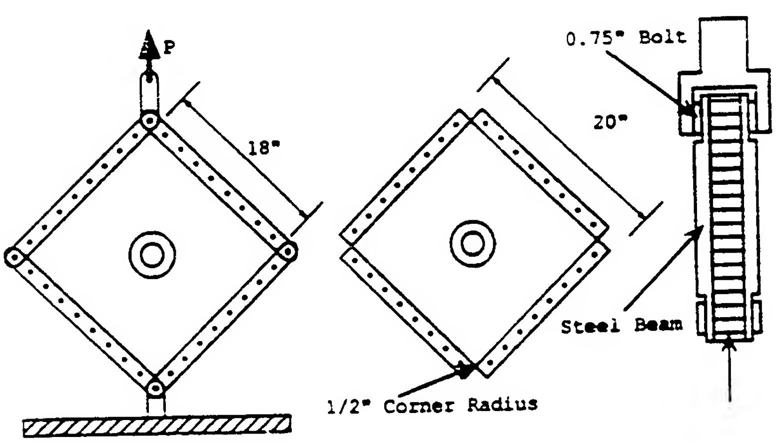


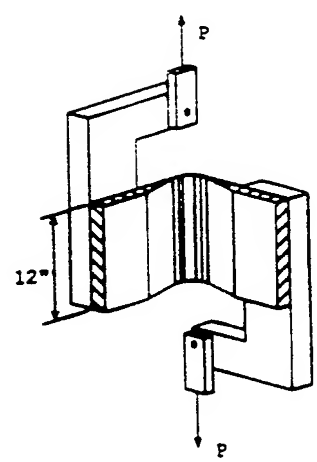






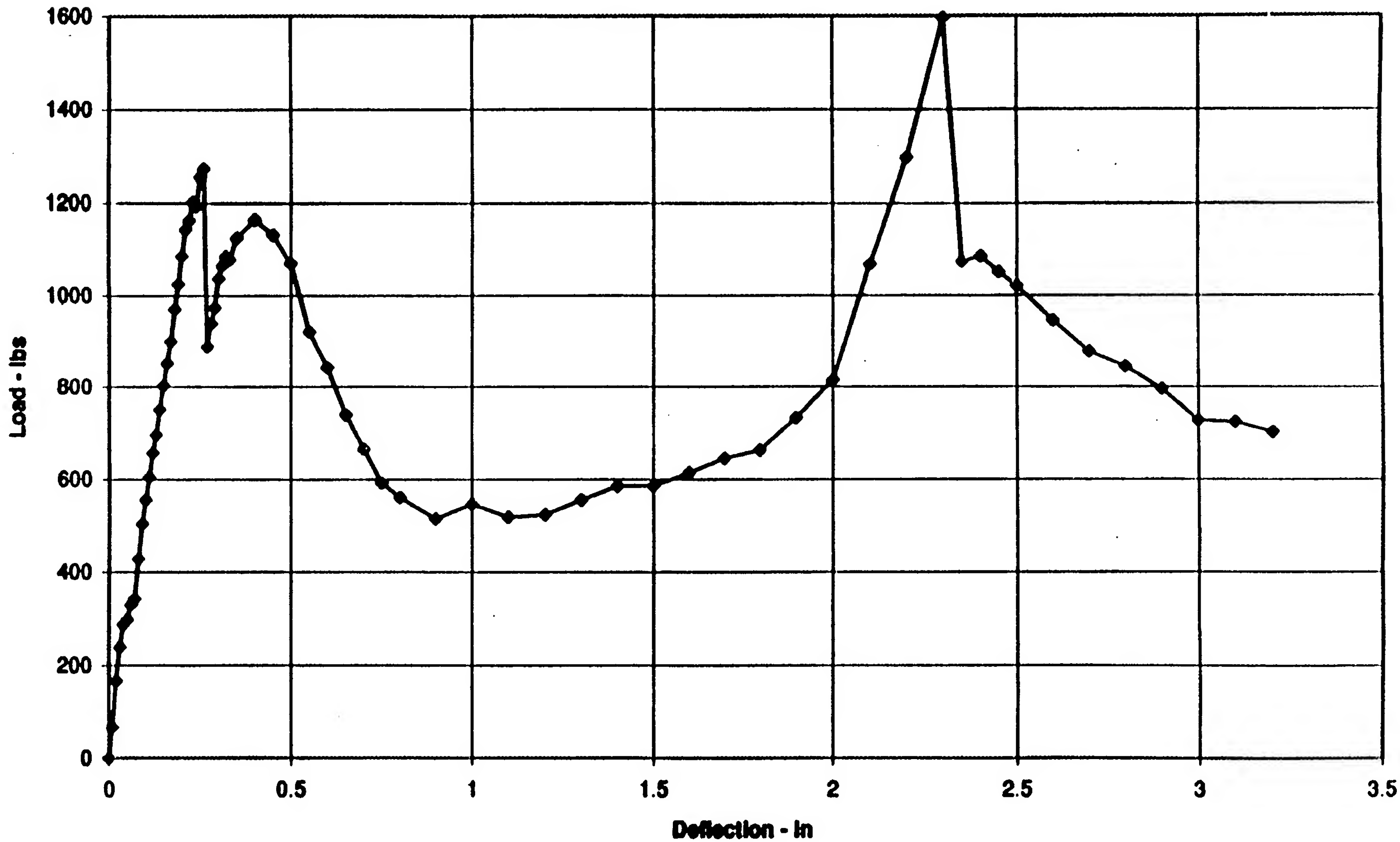


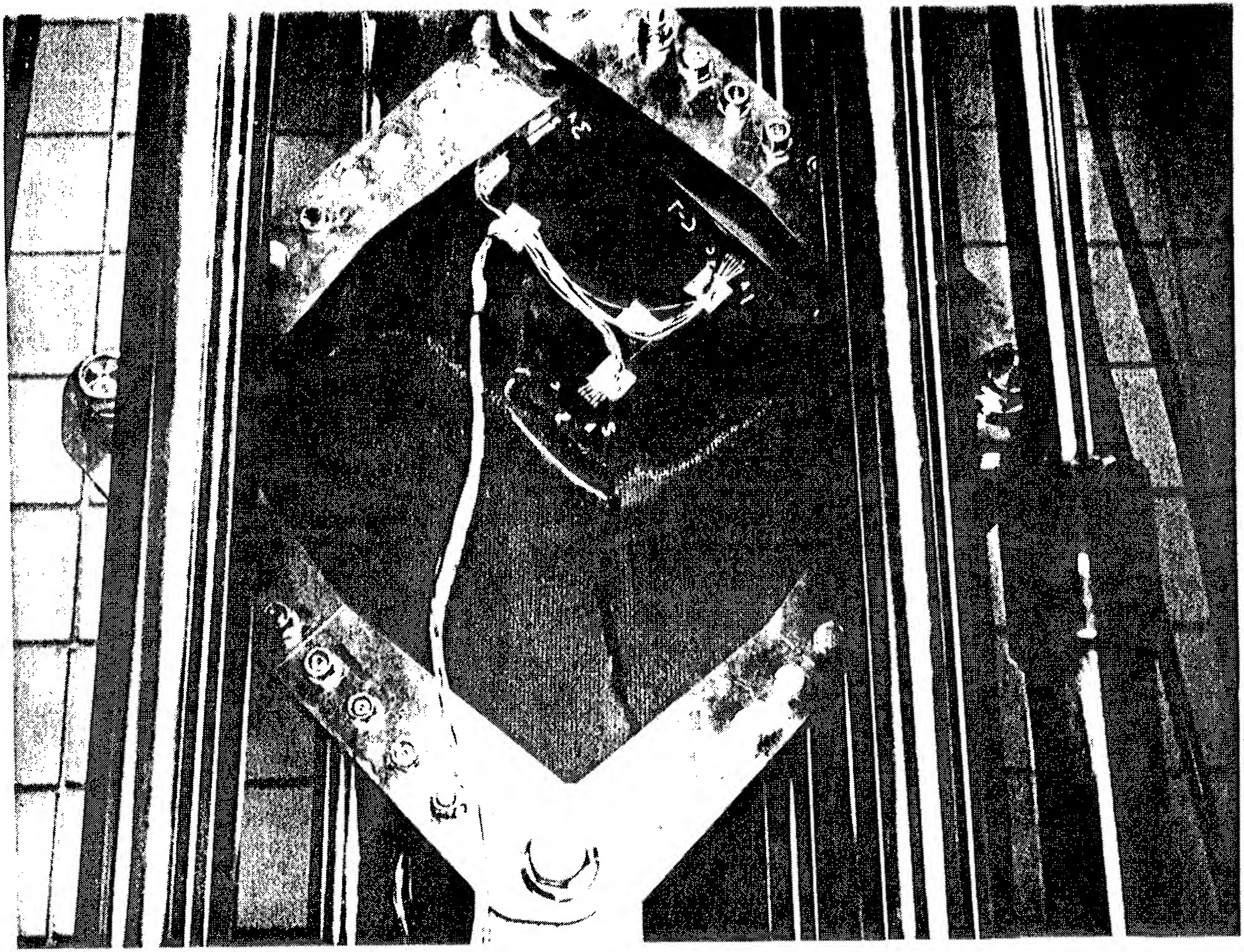




SPONSON REPAIR TEST SPECIMENS

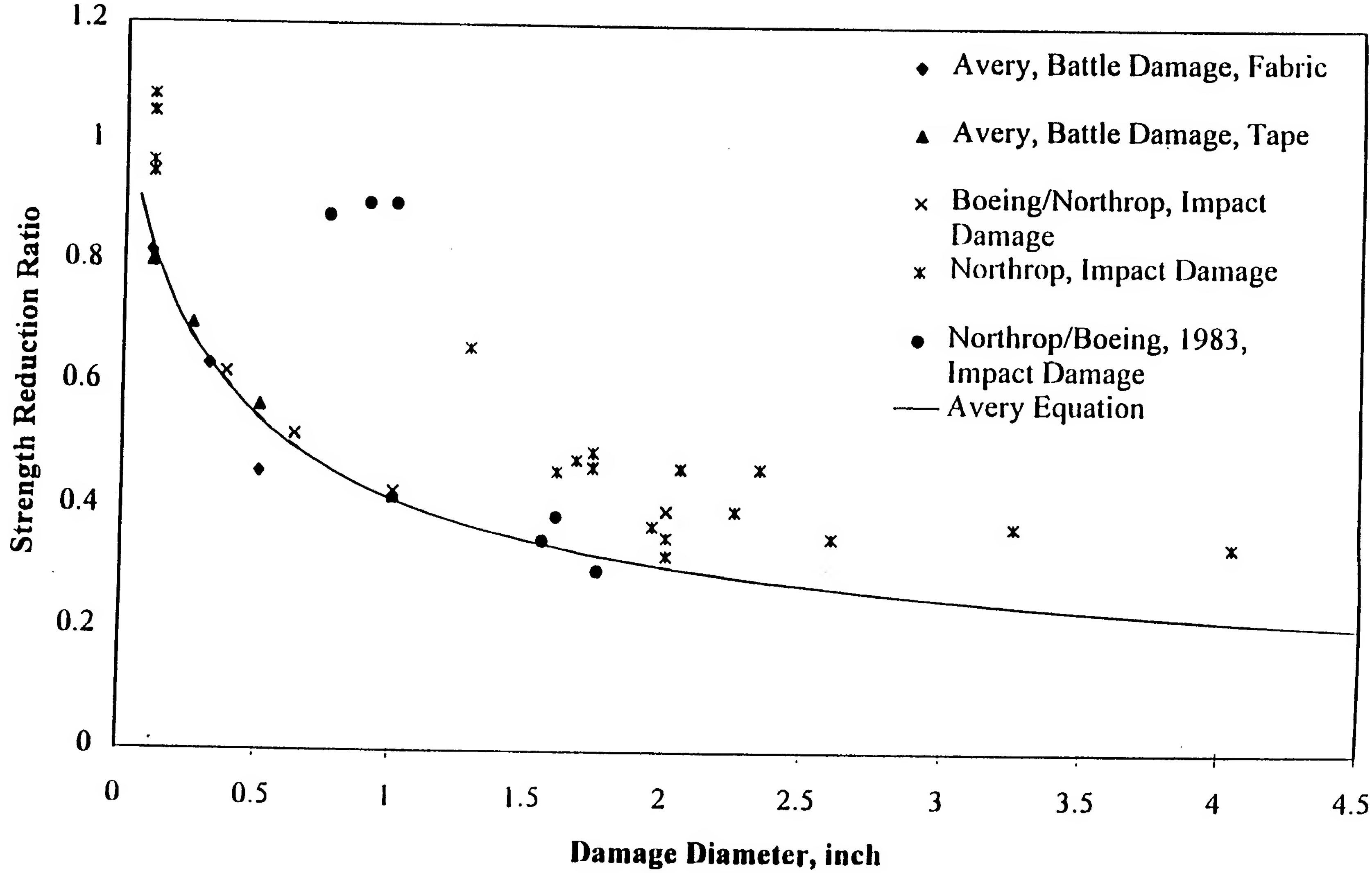
Penetration - 2" Sandwich Beam - 4" Fork Lift



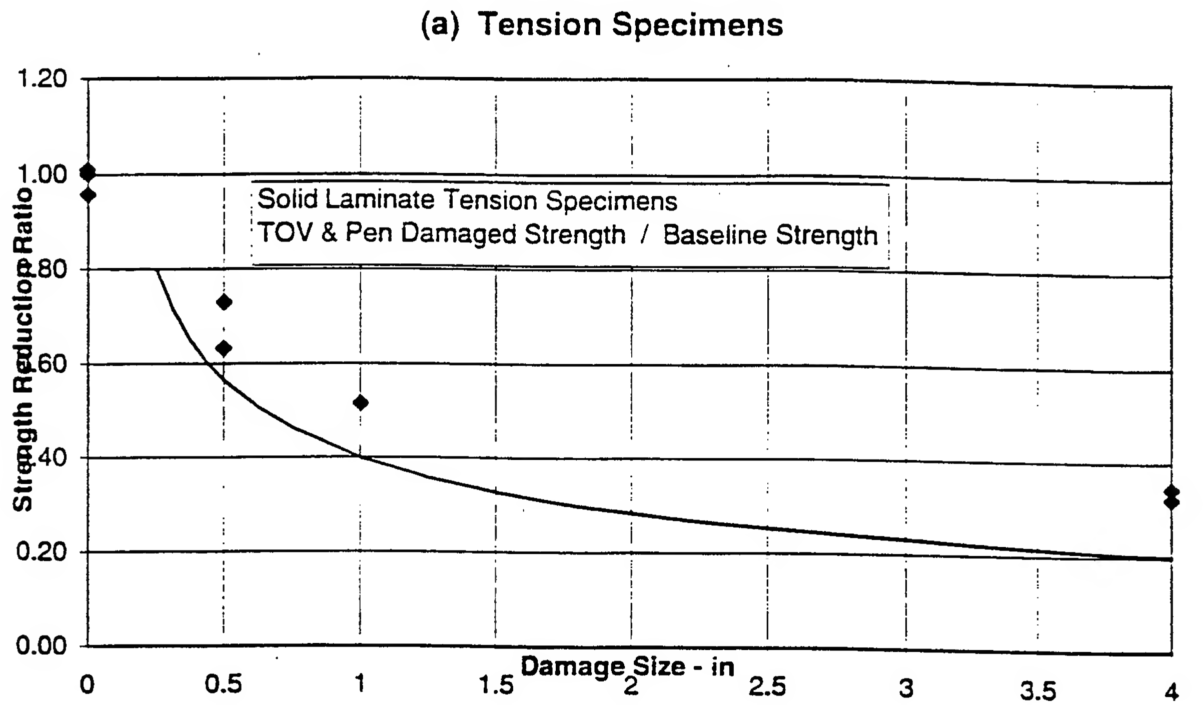


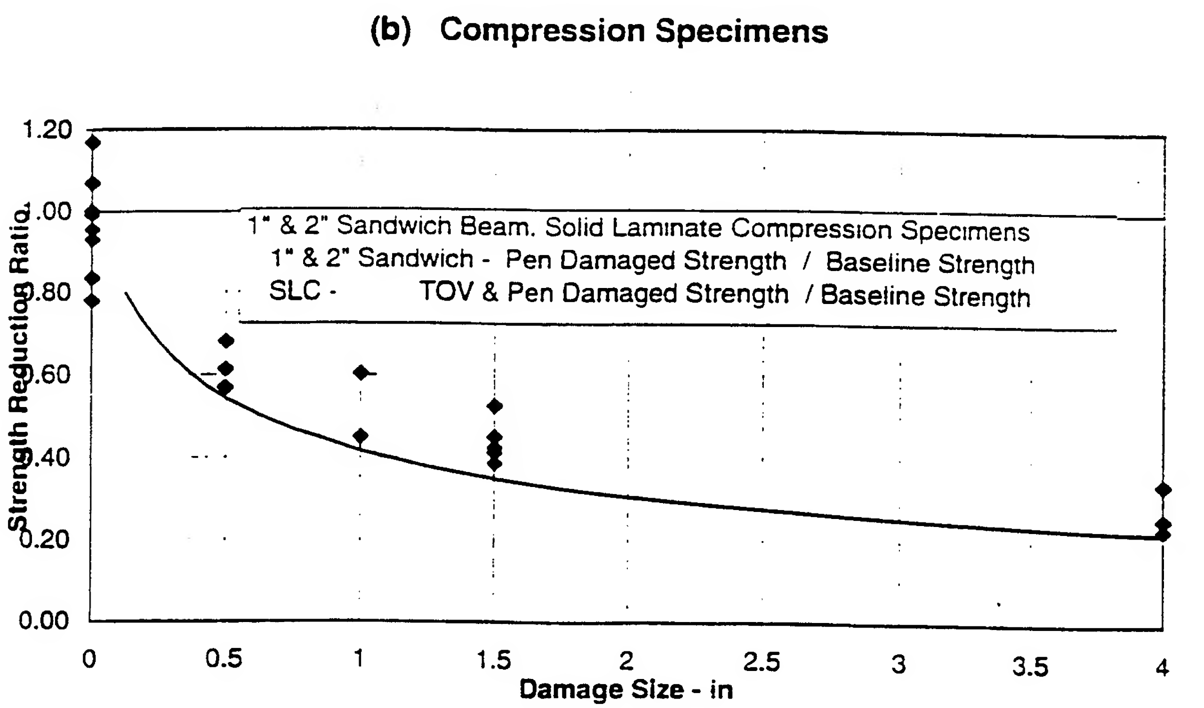
SANDWICH SHEAR TEST - FORK LIFT PENETRATION DAMAGE

STRENGTH REDUCTION AS A FUNCTION OF DAMAGE SIZE - COMPRESSION

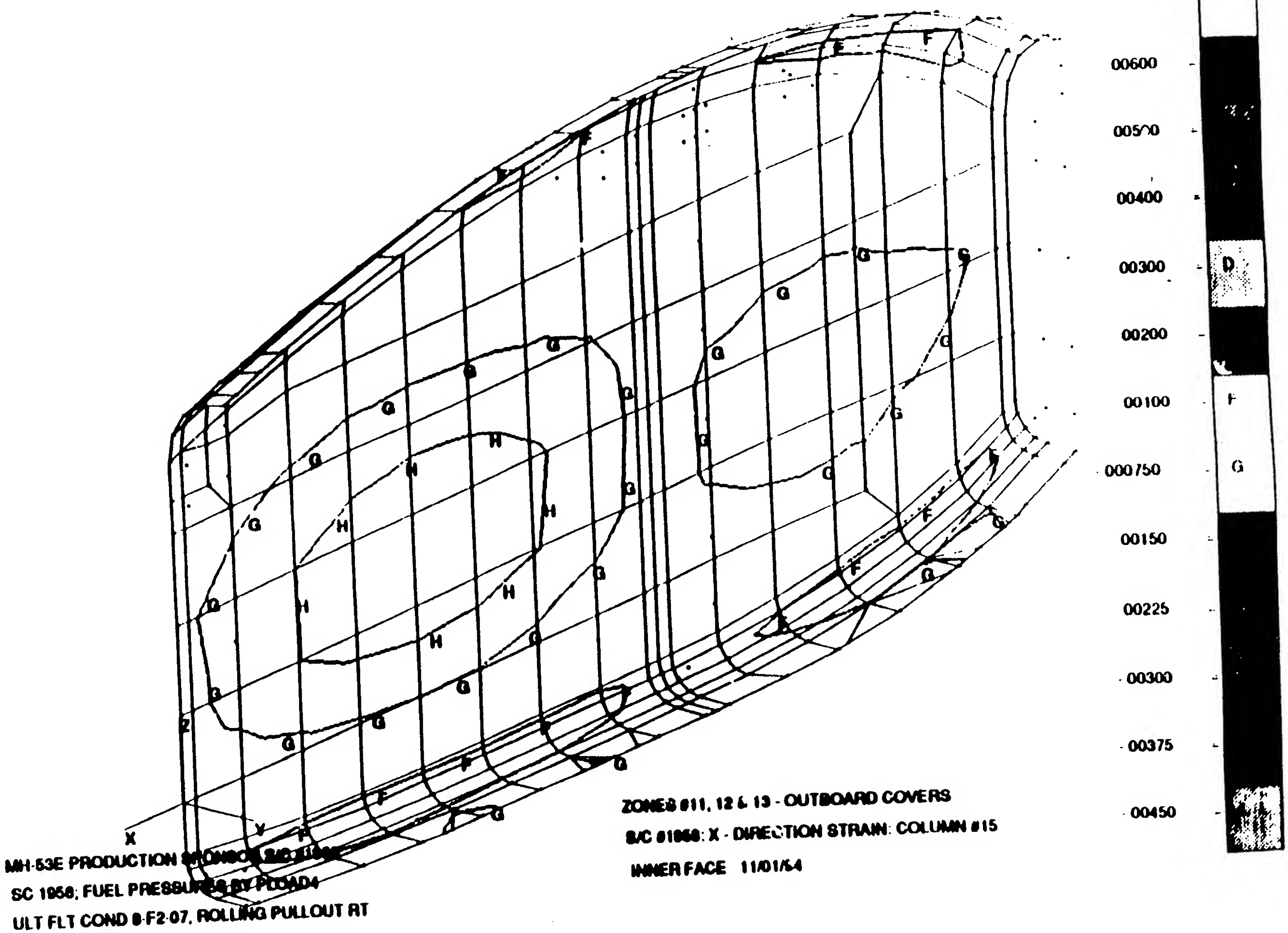


SPONSON PROGRAM SPECIMEN STRENGTHS AS A FUNCTION OF DAMAGE SIZE



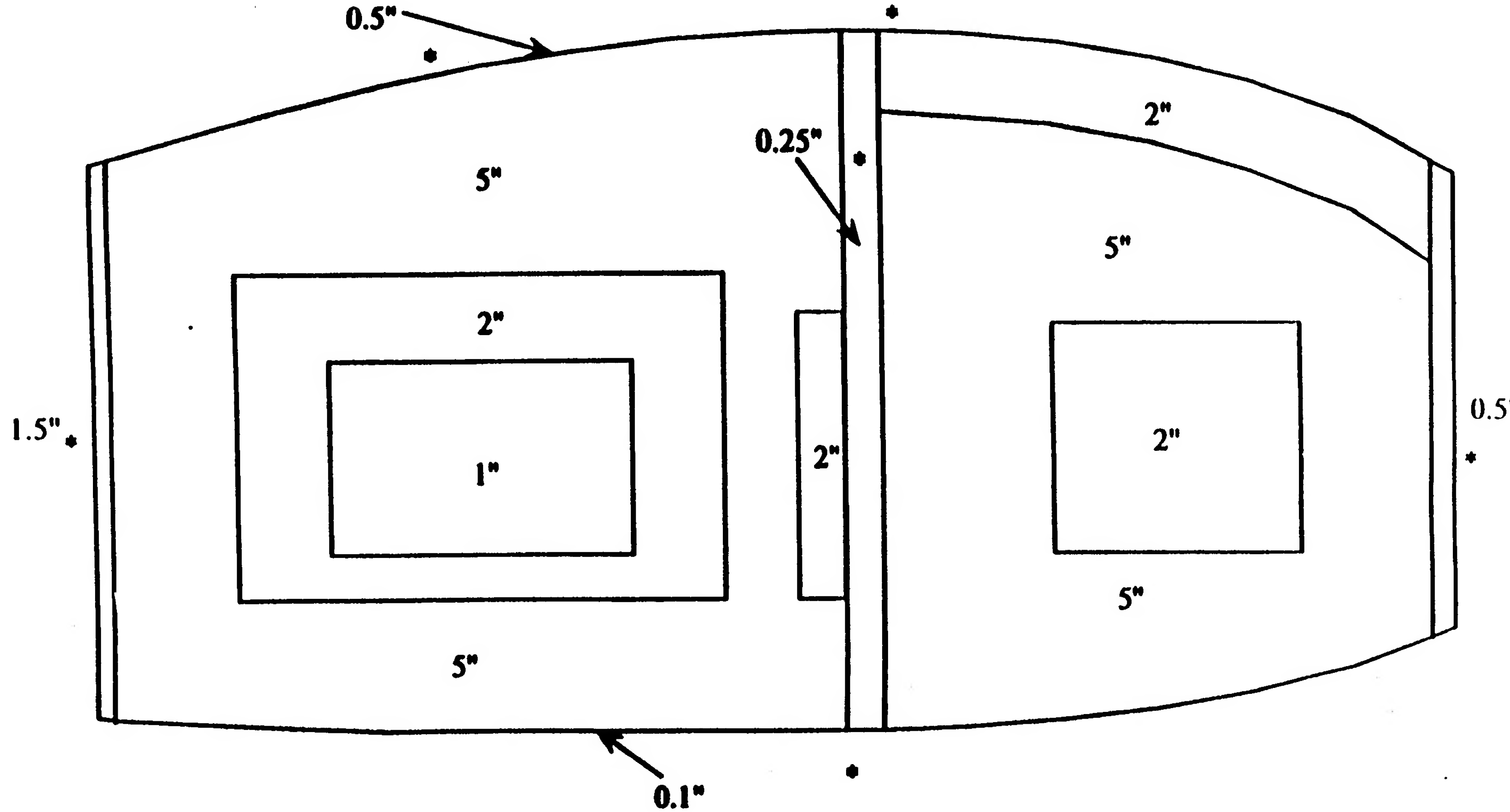


STRAIN MAP - OUTER COVER



ALLOWABLE DAMAGE PLOT - OUTER COVER

Outboard Cover Panel, Outer Facesheet

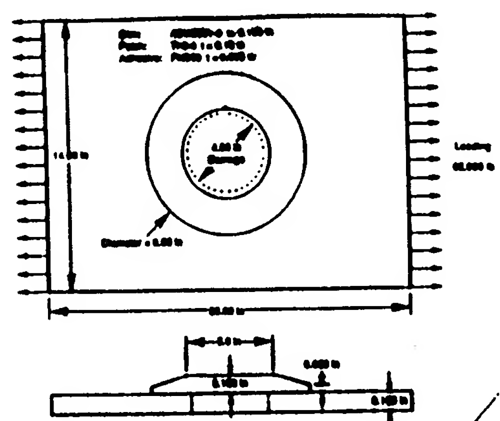


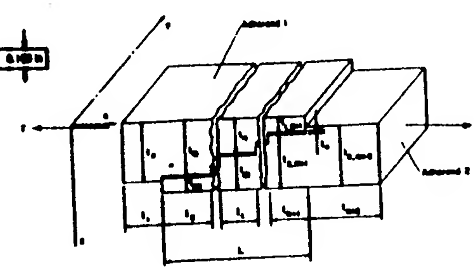
* Attachment Region

REPAIR ANALYSIS METHODS

BONDED REPAIRS

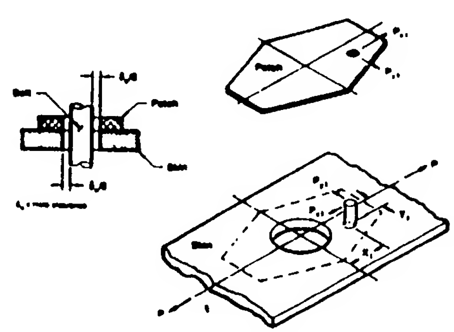
- PGLUE
- A4EI
- ESDU 8039
- Y004





BOLTED REPAIRS

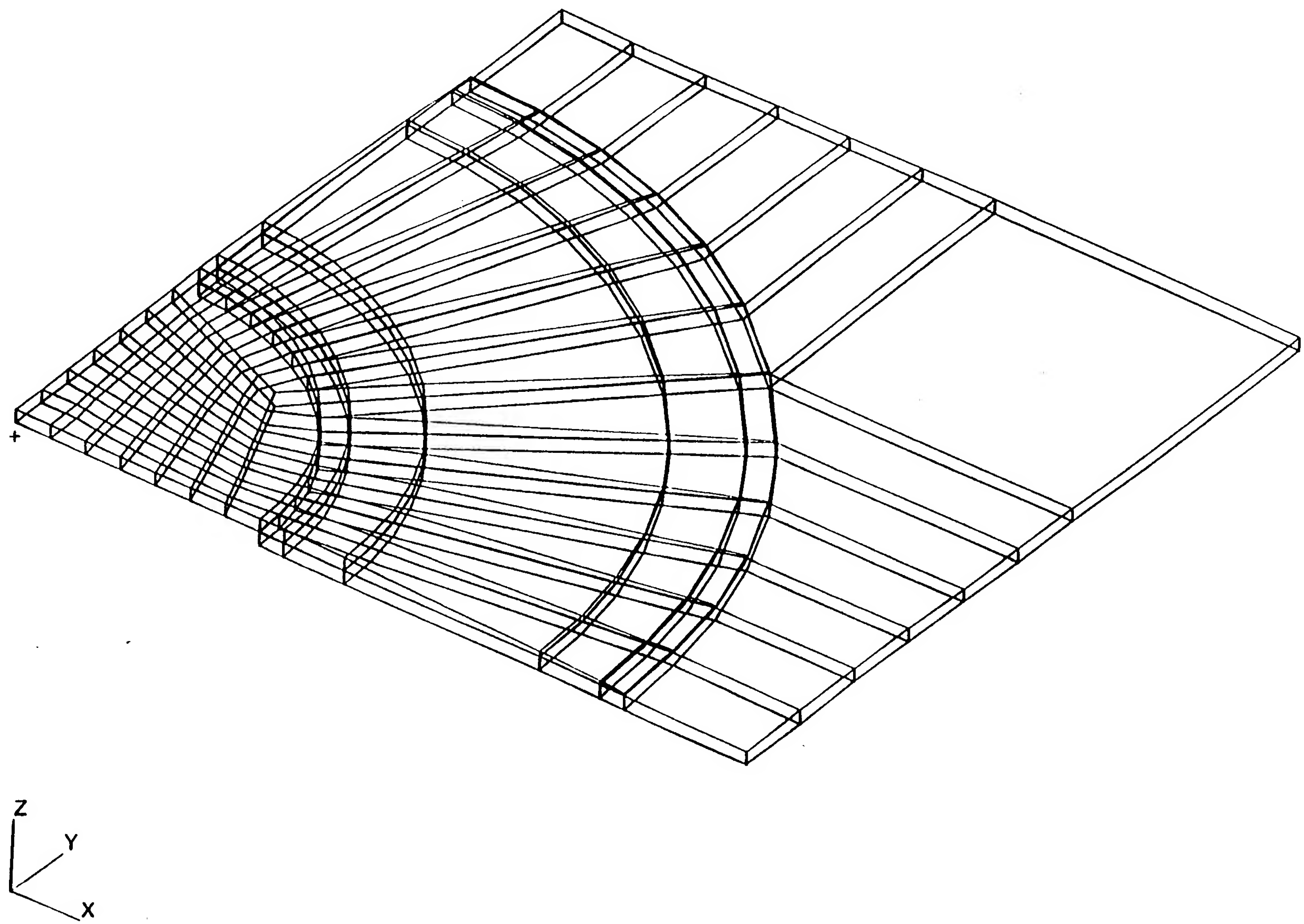
- CREPAIR
- A4EJ
- BJSFM
- BLDM



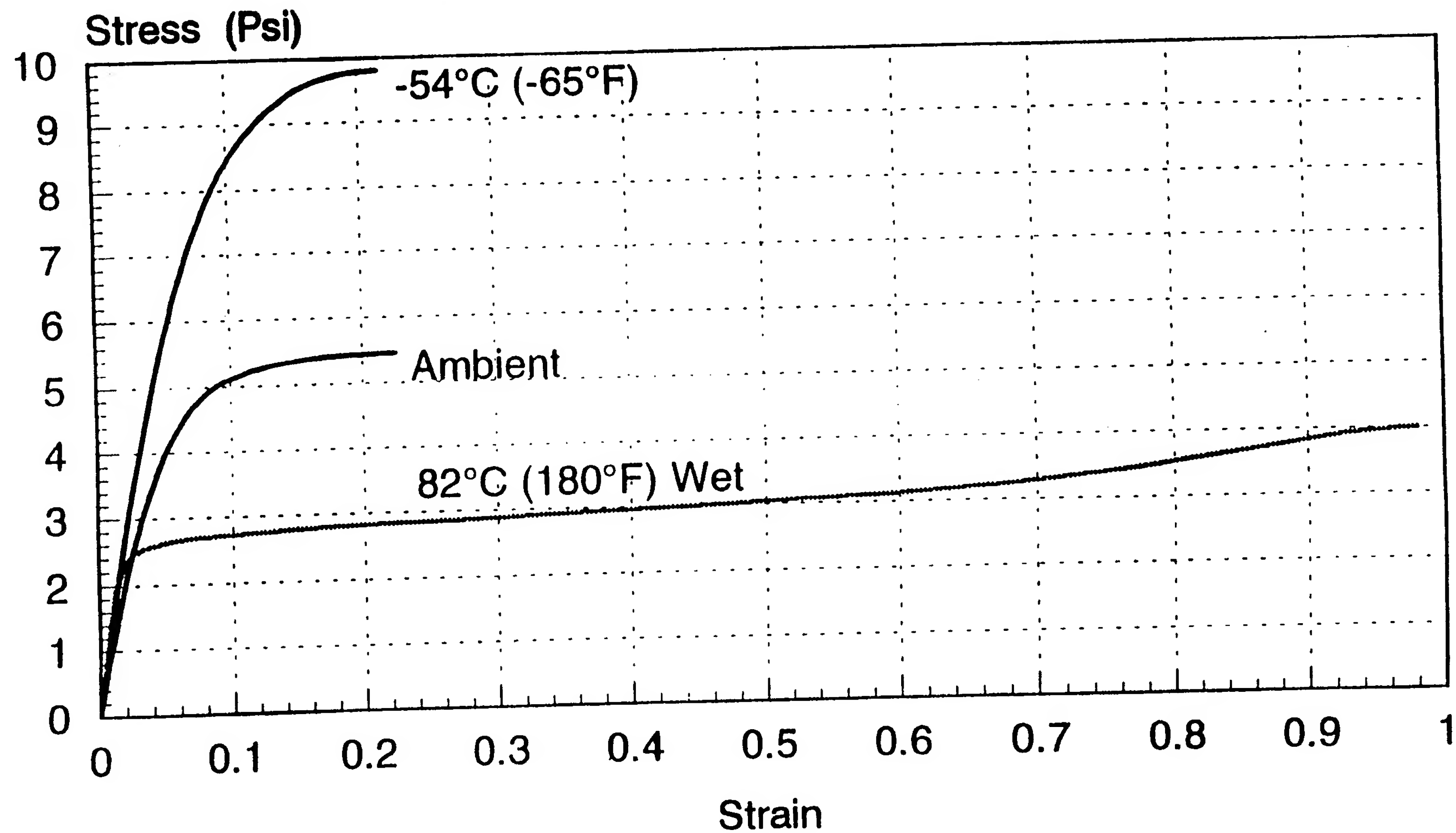
GENERAL PURPOSE

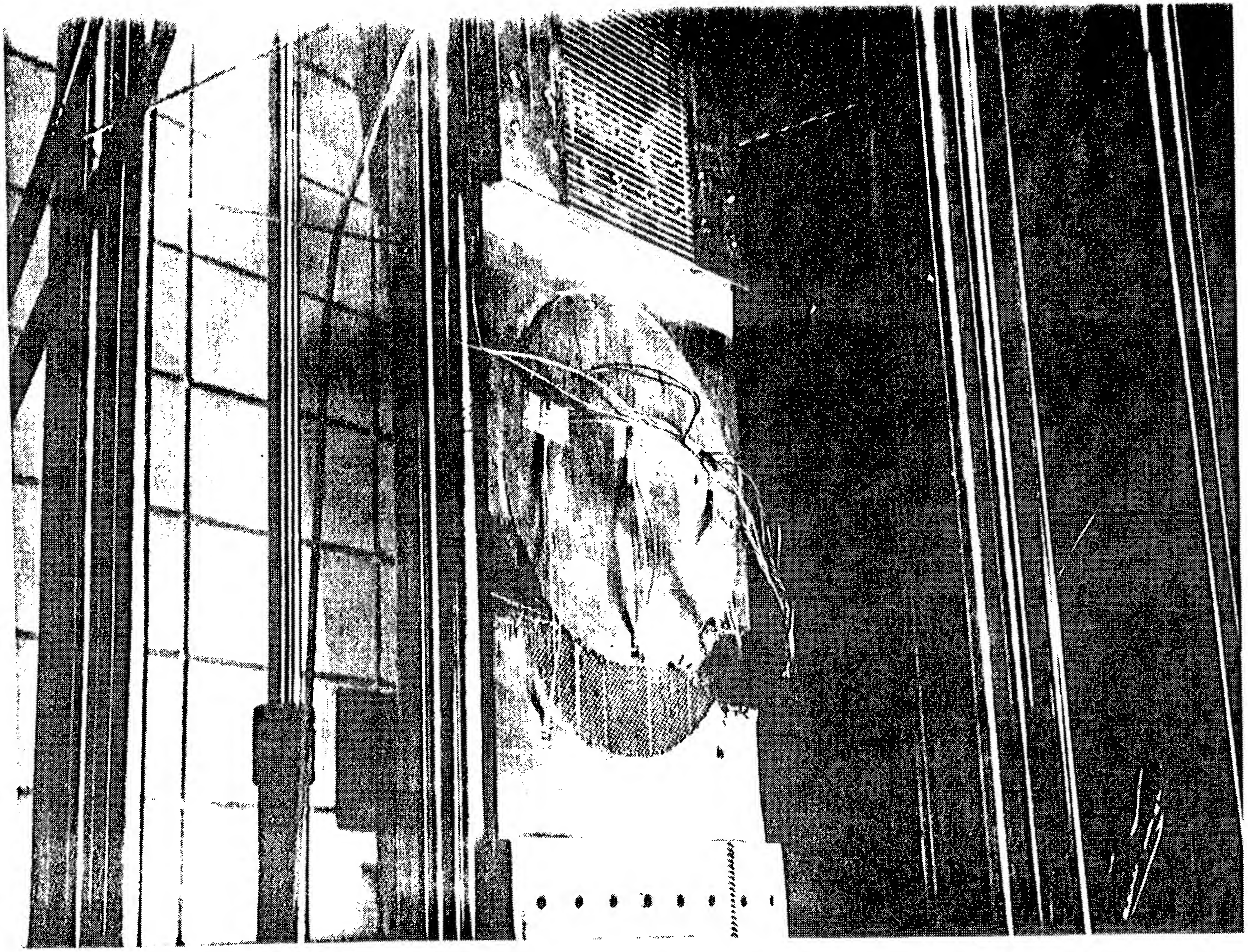
- NASTRAN

PGLUE MODEL OF SOLID LAMINATE TENSION SPECIMEN REPAIR

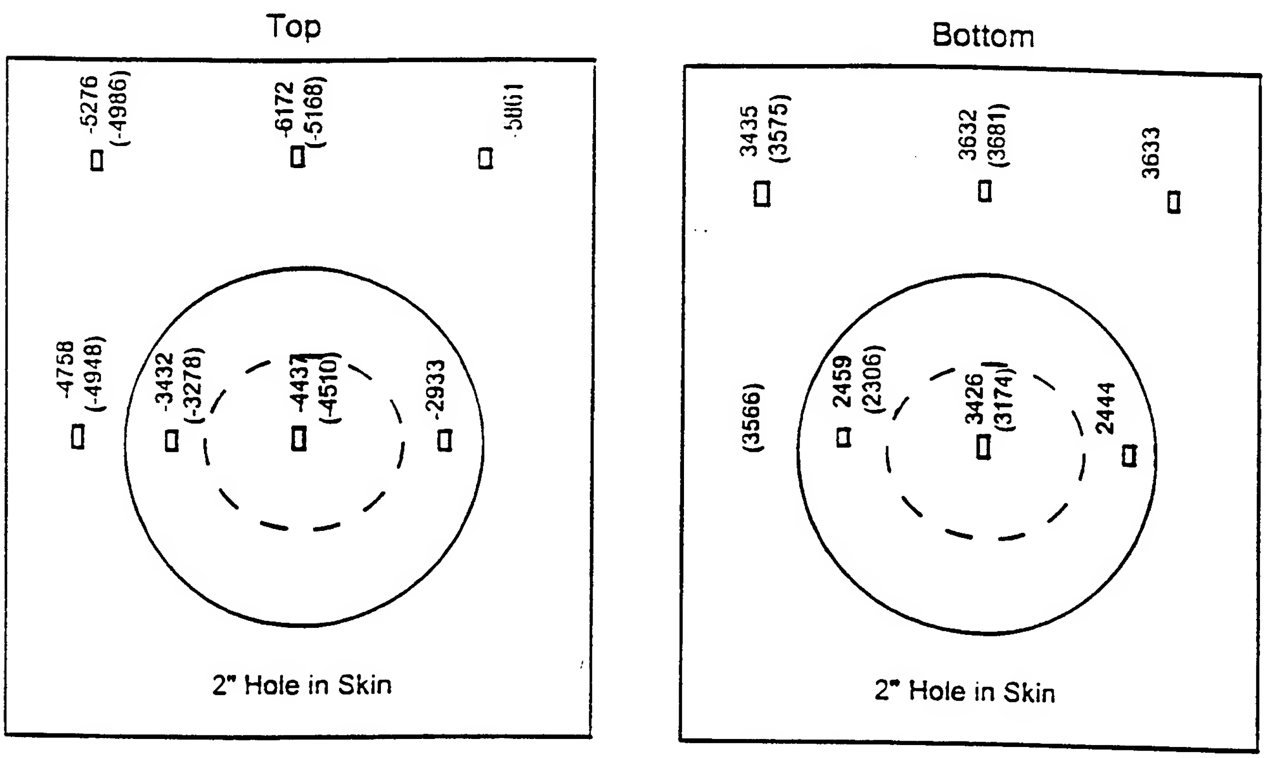


Magnobond 6363



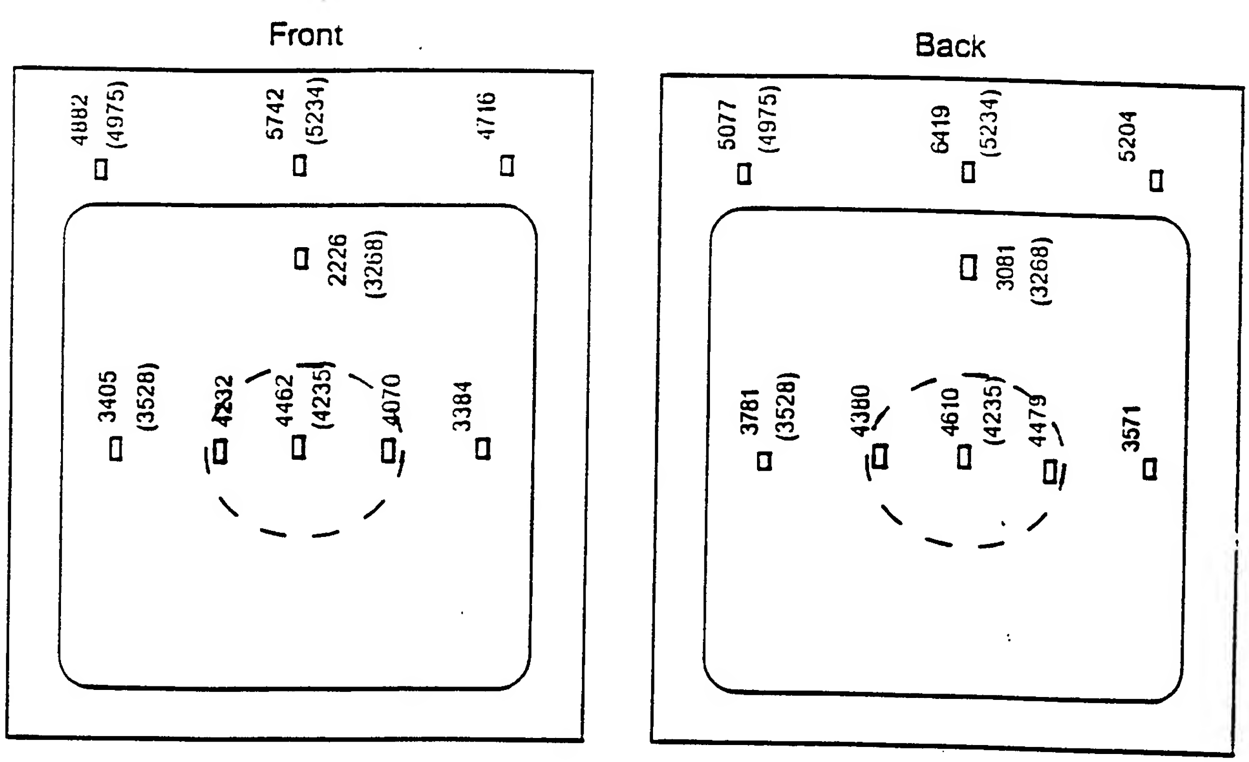


12" SOLID LAMINATE TENSION TEST - 2 SIDED REPAIR



TEST-ANALYSIS CORRELATION - SANDWICH BEAM

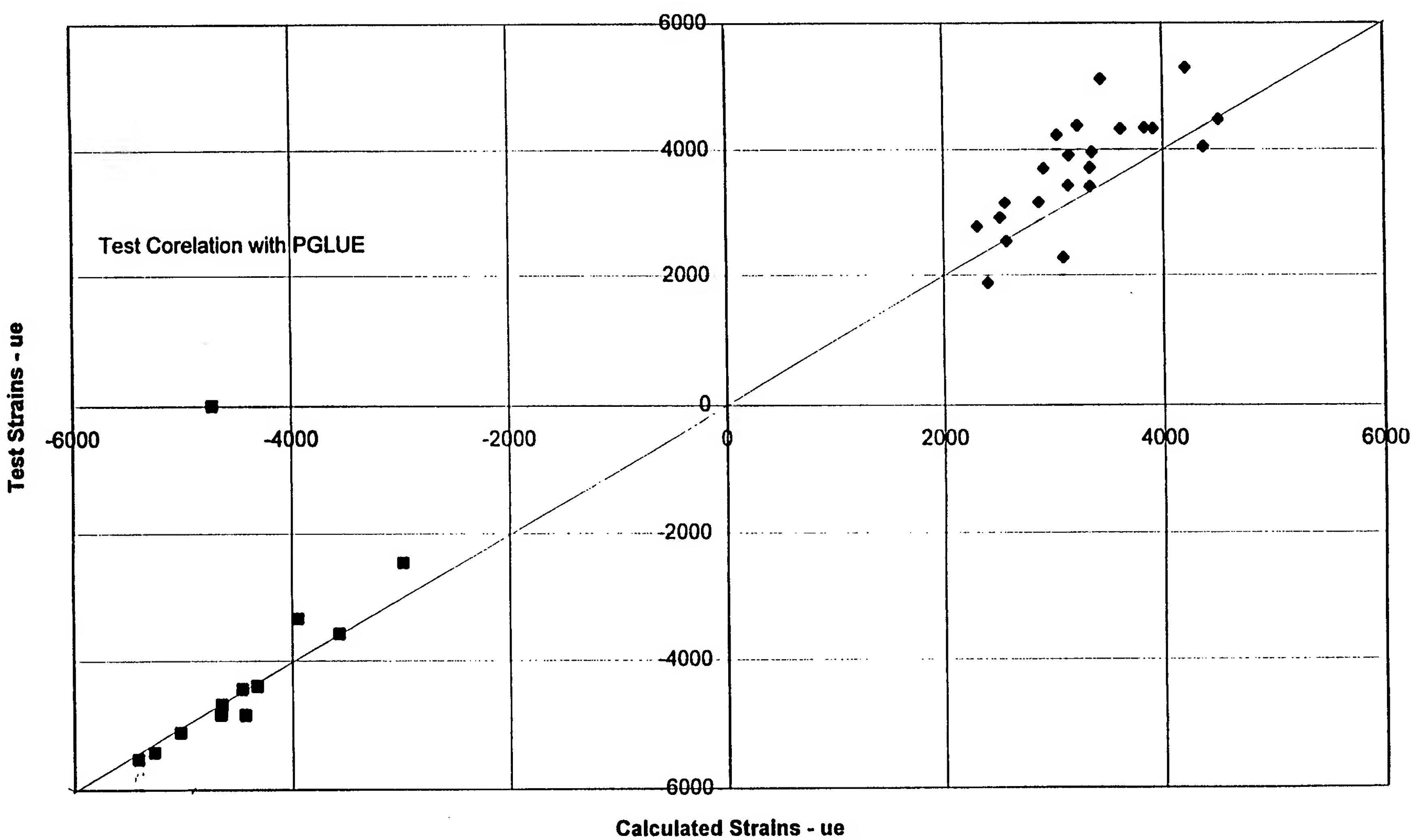
1111 = TEST
(1111) = PGLUE



TEST-ANALYSIS CORRELATION - SOLID LAMINATE

1111=TEST

SANDWICH BEAM PATCH STRAIN CORRELATION PLOT



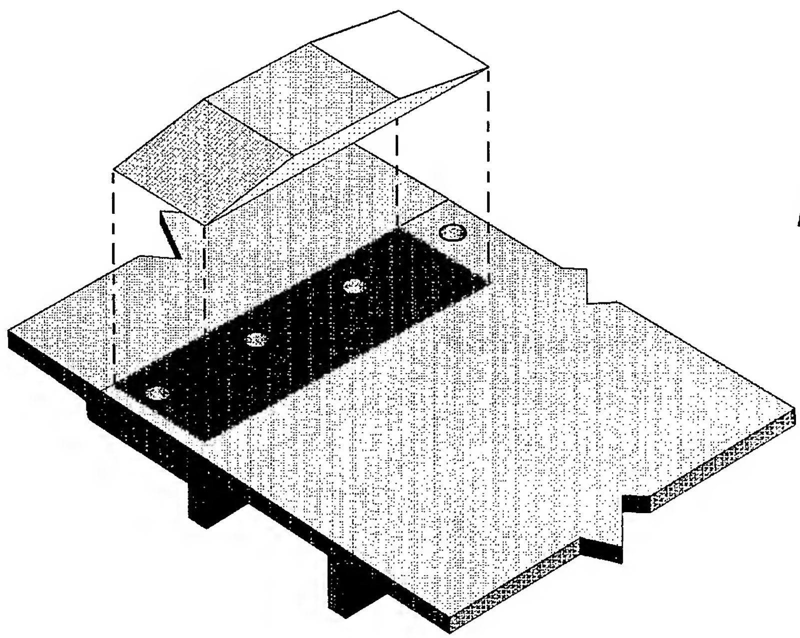
SPONSON REPAIR PROGRAM SUMMARY

- 243 SPECIMENS FABRICATED
- 29 DAMAGED SPECIMENS SECTIONED
- 40 BASELINE SPECIMENS TESTED
- 117 DAMAGED SPECIMENS TESTED
- 98 REPAIRED SPECIMENS TESTED

- REPAIR METHODOLOGY VALIDATED
  - * DOUBLE VACUUM DEBULK SUCCESSFUL
  - * USE PATCH THICKNESS SAME AS BASE MATL
  - * NEED BETTER PASTE ADHESIVE
  - * PGLUE ANALYSIS OK FOR IN-PLANE STRAINS, NOT AS GOOD FOR ADHESIVE SHEAR STRAINS

- SRM IN DEVELOPMENT

Putting FAILSAFETY Back Into Aircraft Structures



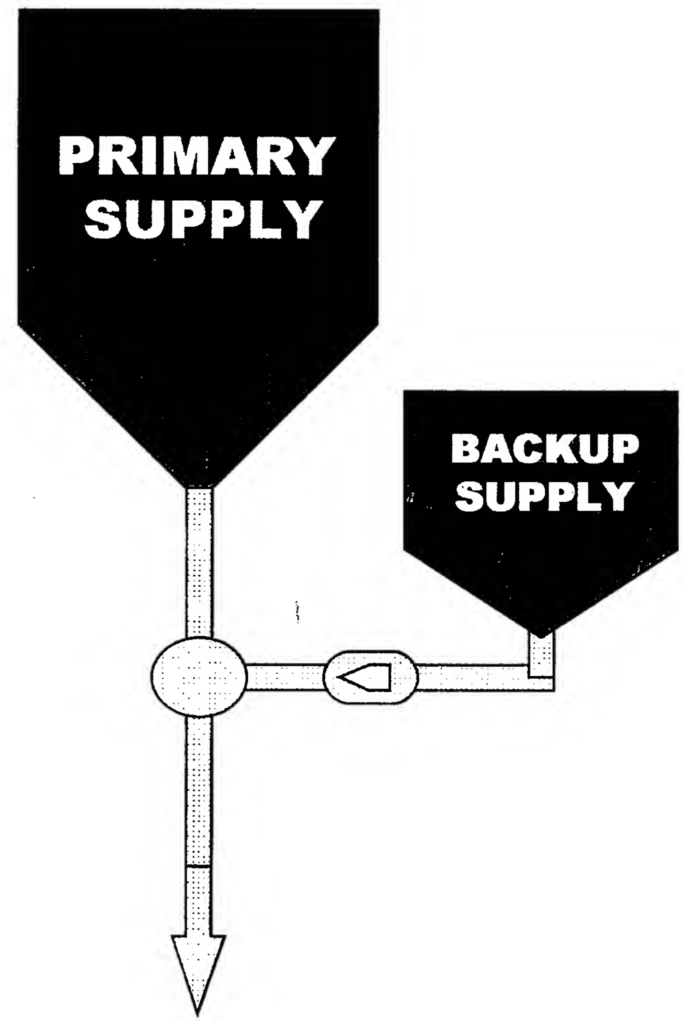
One Ridgmar Centre, 6500 West Freeway
Fort Worth, Texas 76116
(817) 737-1655

Alan Kerr
817-737-1680
kerra@dyncorp.com

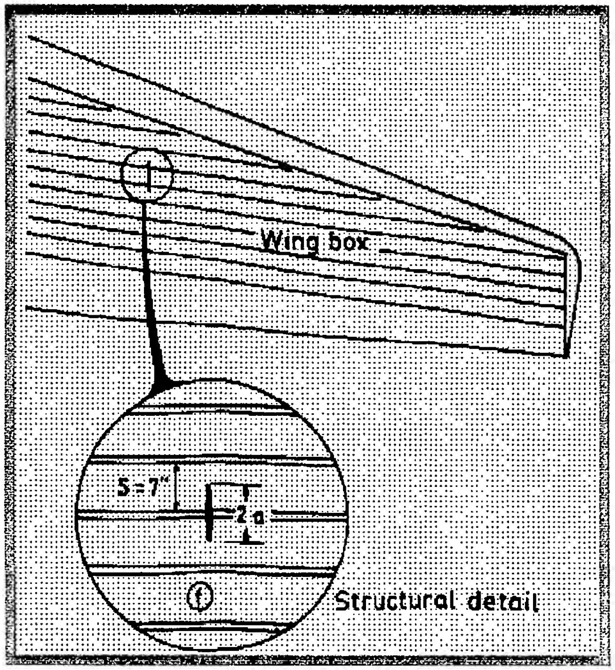
The Bonded Repair Company

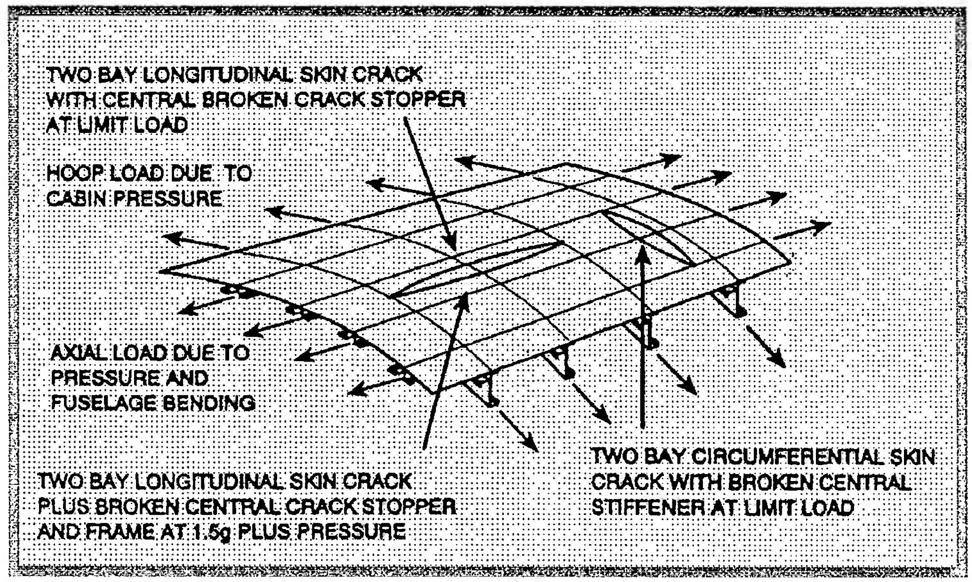
What is Failsafety?

- A Method For Fatigue Life Management
- Required That
  - the "catastrophic failure or excessive ......, are not probable after fatigue failure, or obvious partial failure of a single principal structural element.
    After such failure, the remaining structure shall be capable of withstanding static load corresponding to .... (*range of flight conditions*)..."
- A Method of Achieving Safety



Failsafety Demonstration





- **Static Load Tests Done For Type Certification**
- **Demonstrated Residual Strength In the Presence of Large Defects**
- **Provides a Comfortable Feeling**

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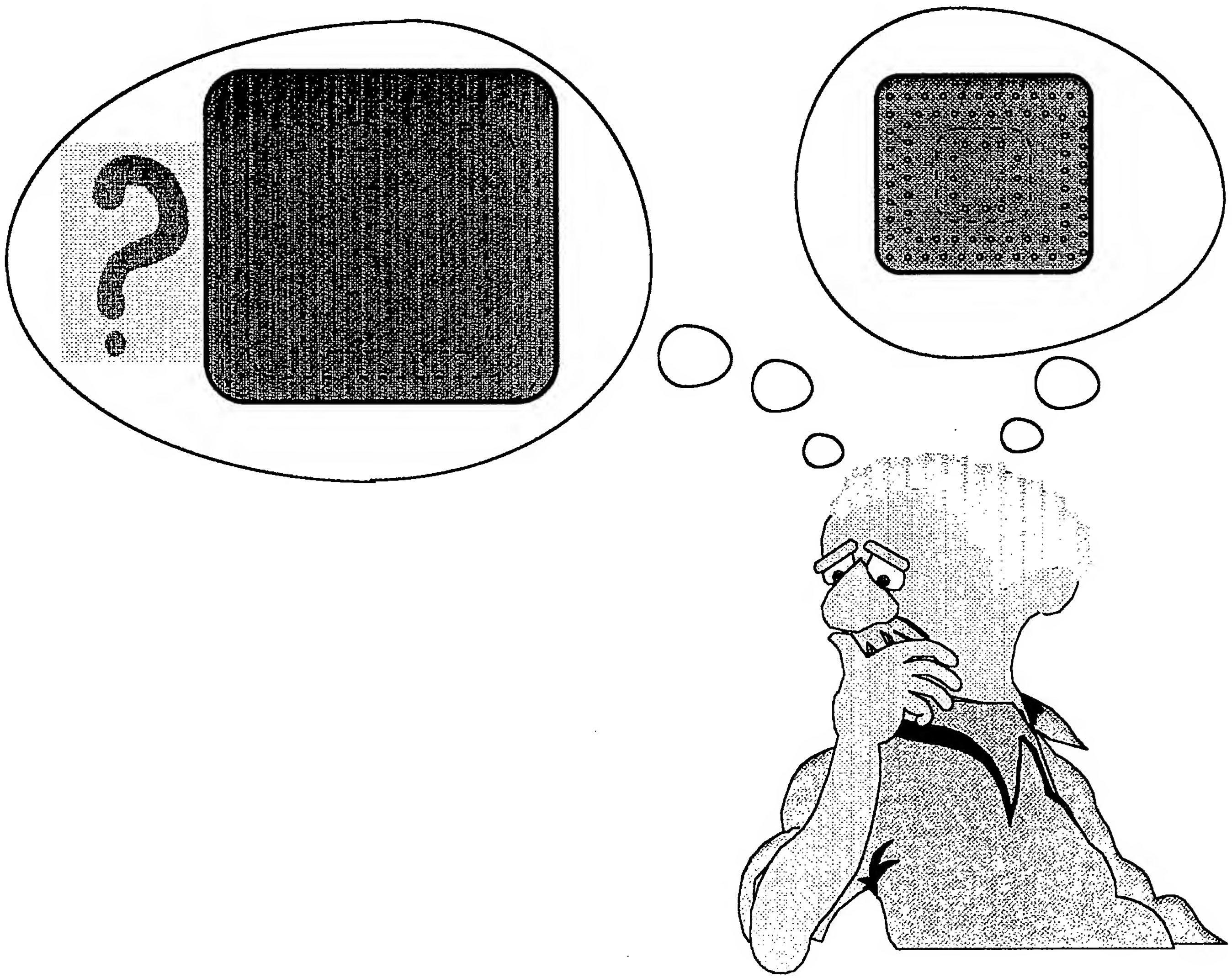
What's The Concern?



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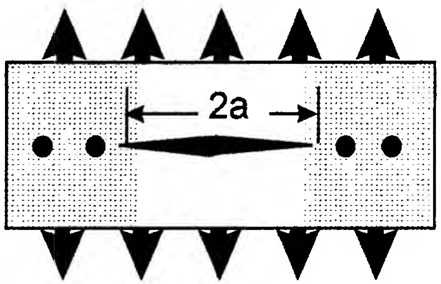
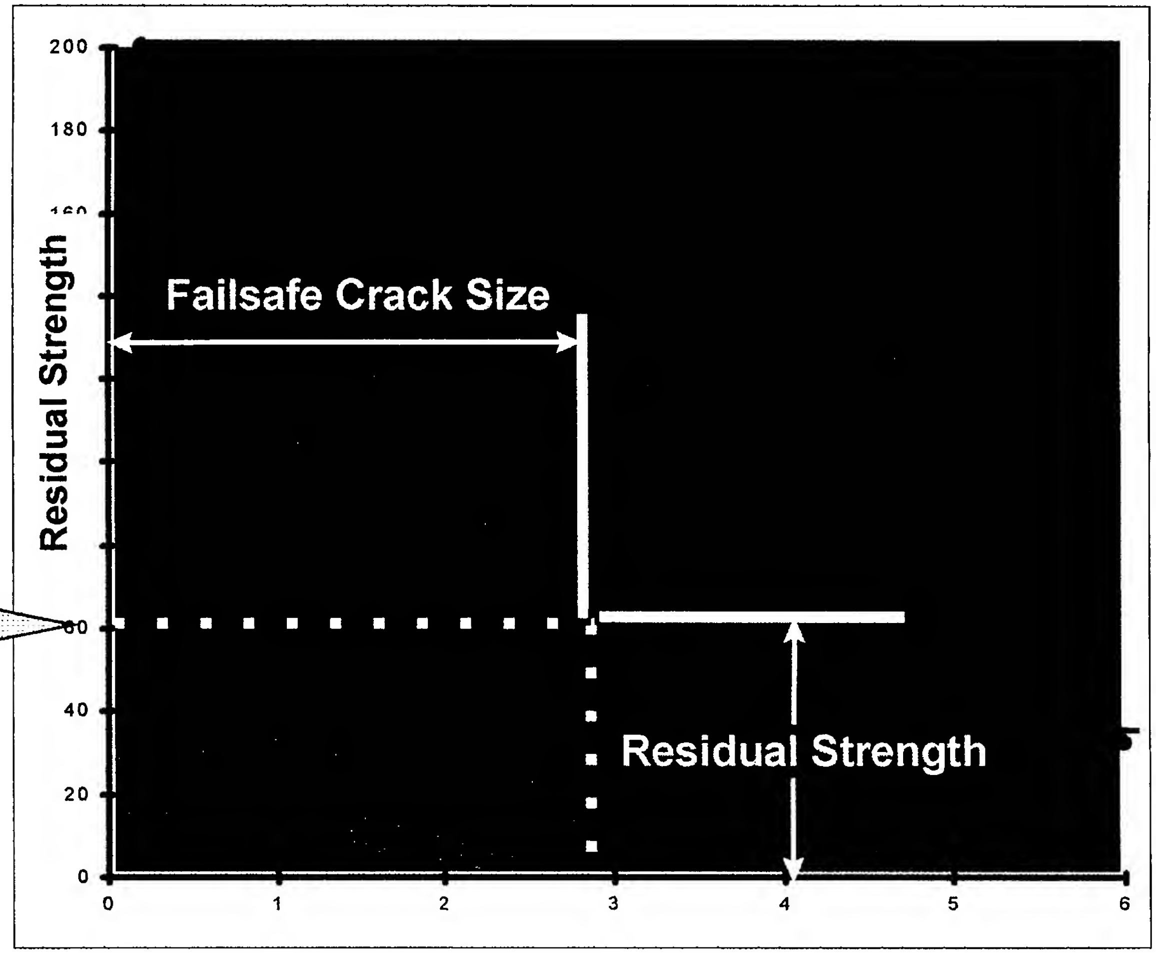
Residual Strength Curves
Undamaged Unstiffened Panel

No Damage





- Basis For Original Aircraft Certification

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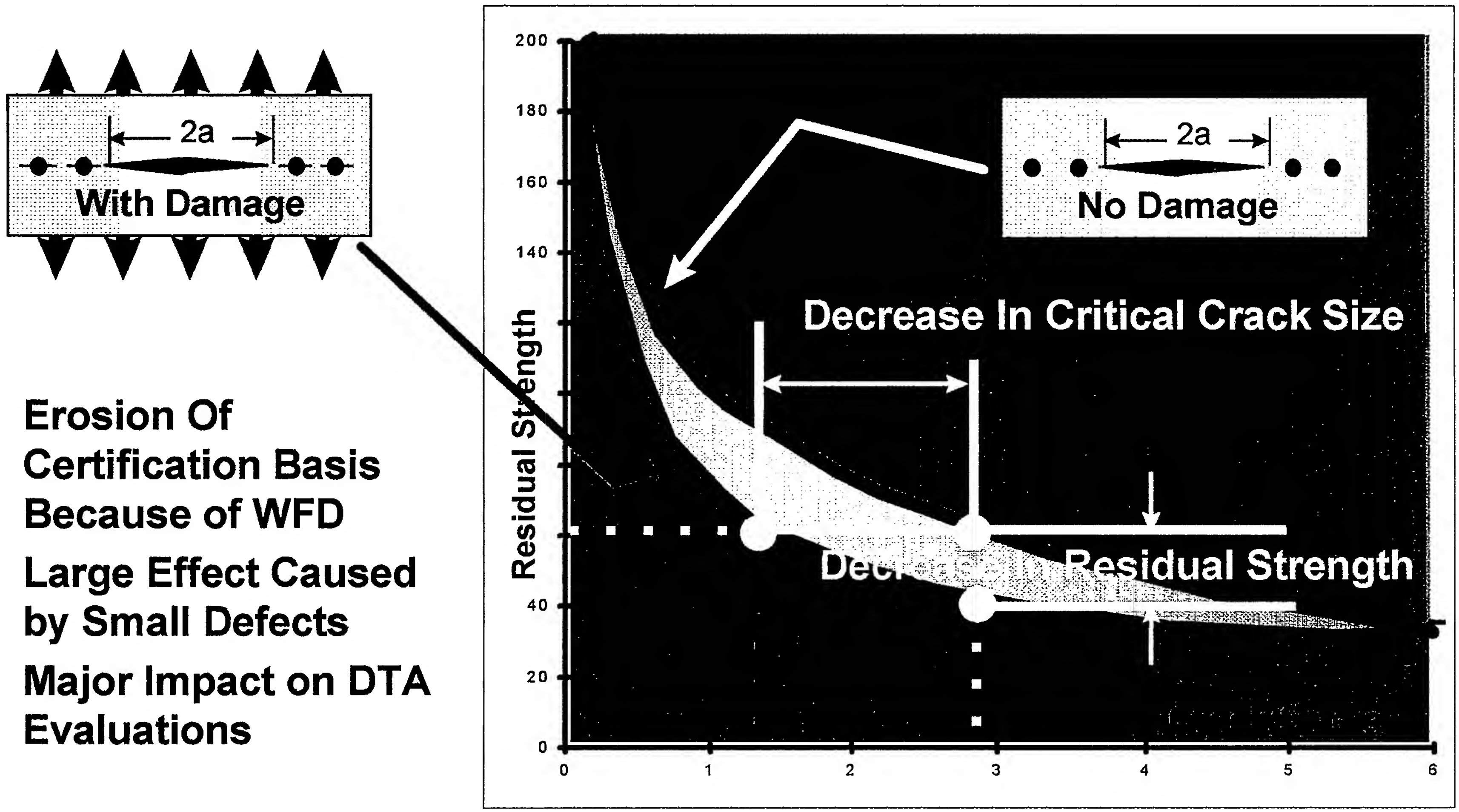
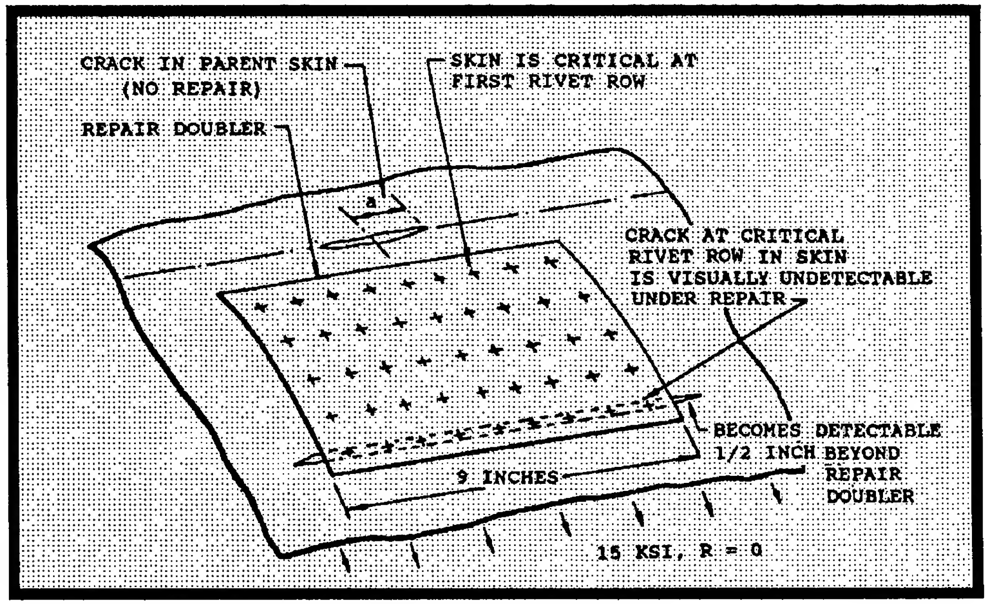
Residual Strength Curves
WFD Damaged Unstiffened Panel



- Erosion Of Certification Basis Because of WFD
- Large Effect Caused by Small Defects
- Major Impact on DTA Evaluations

Fuselage Lap Joints

- A Growing Problem
- Management By Inspection
  - Difficult Because of the Small Defect Detection Requirements
- Conventional Reinforcement Introduces More Problems
- Modification
  - Replacement of Fasteners To Give Improved Fatigue Characteristics
- Panel Replacement
- Repair Cracks



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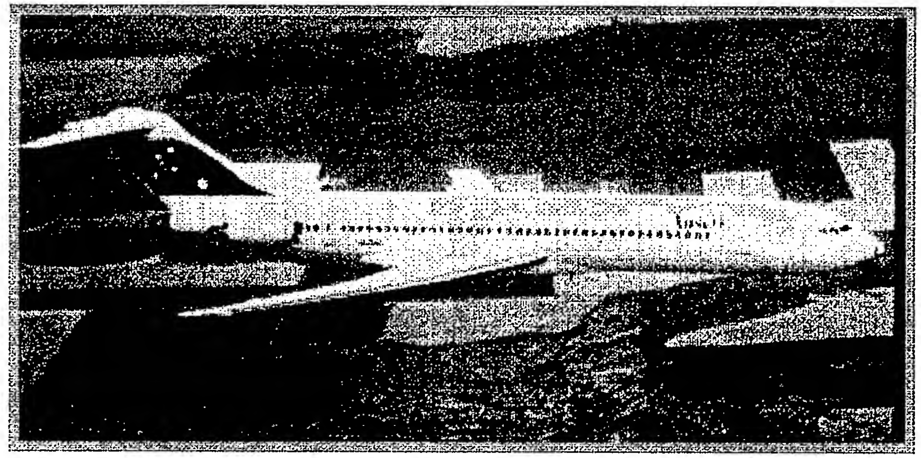
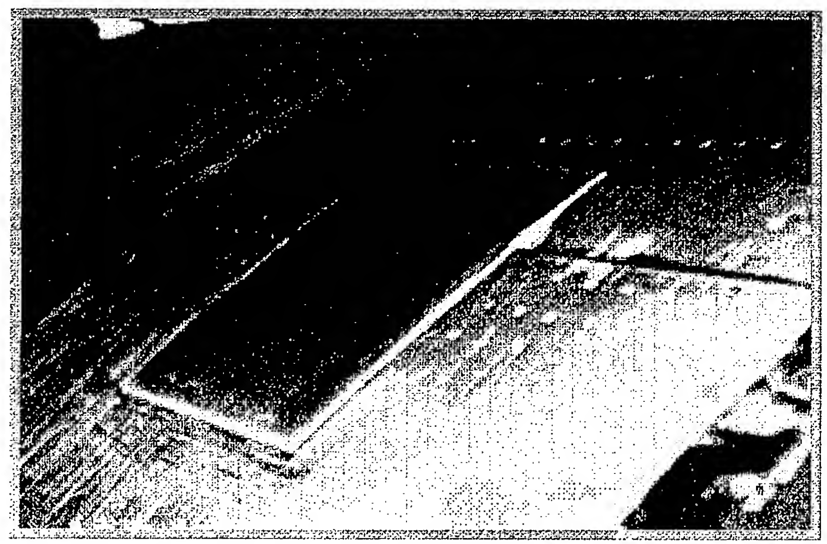
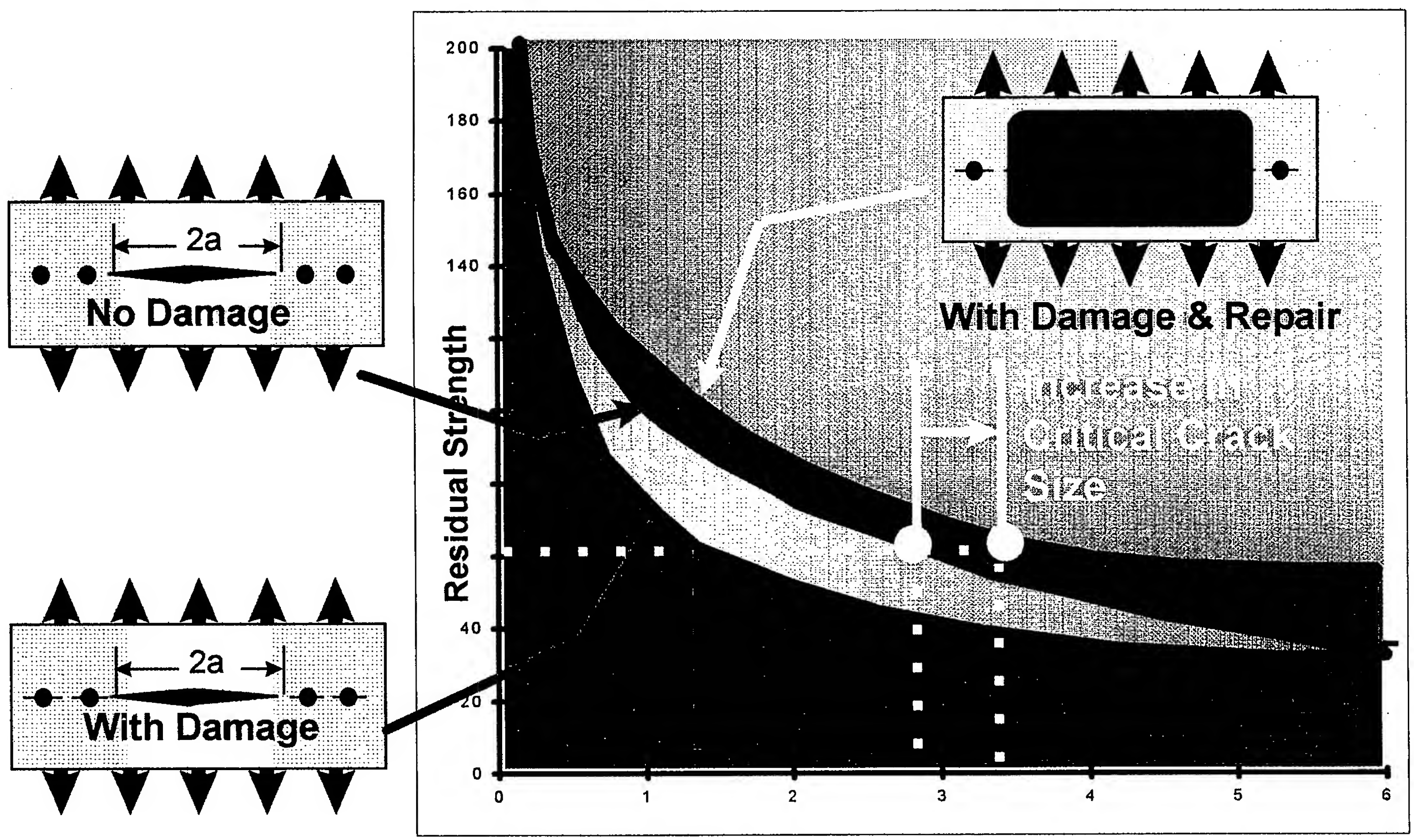
B727 Lap Joint Bonded Repair

- Lap Splice Reinforcement Developed by AMRL as a repair alternative.
- Tests Results
  - Simulated Damage In Lap Splice
  - Restoration of Residual Strength With Damage Present
  - MSD Growth Rate Significantly Reduce
  - Very Long Service Life After Repair





Residual Strength After Repair



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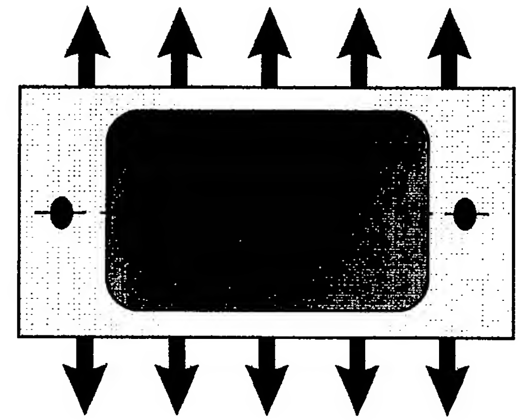
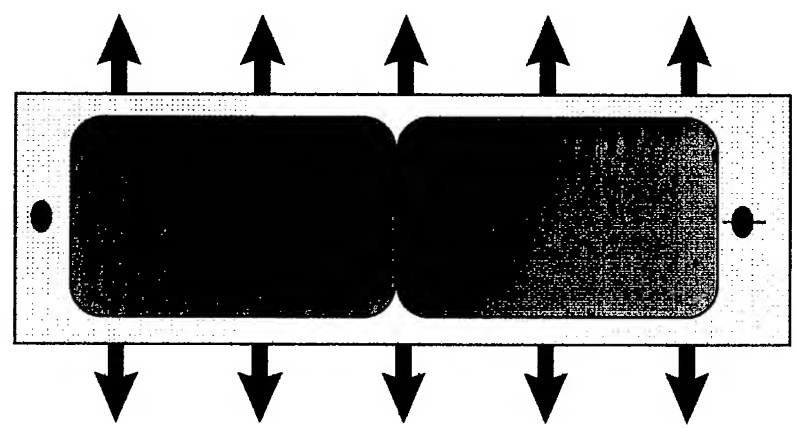
Bonded Reinforcement

- Functions in Several Ways
  - Works as a "Crack Patcher"
  - Restores Residual Capabilities
  - Modifies The Failure Modes

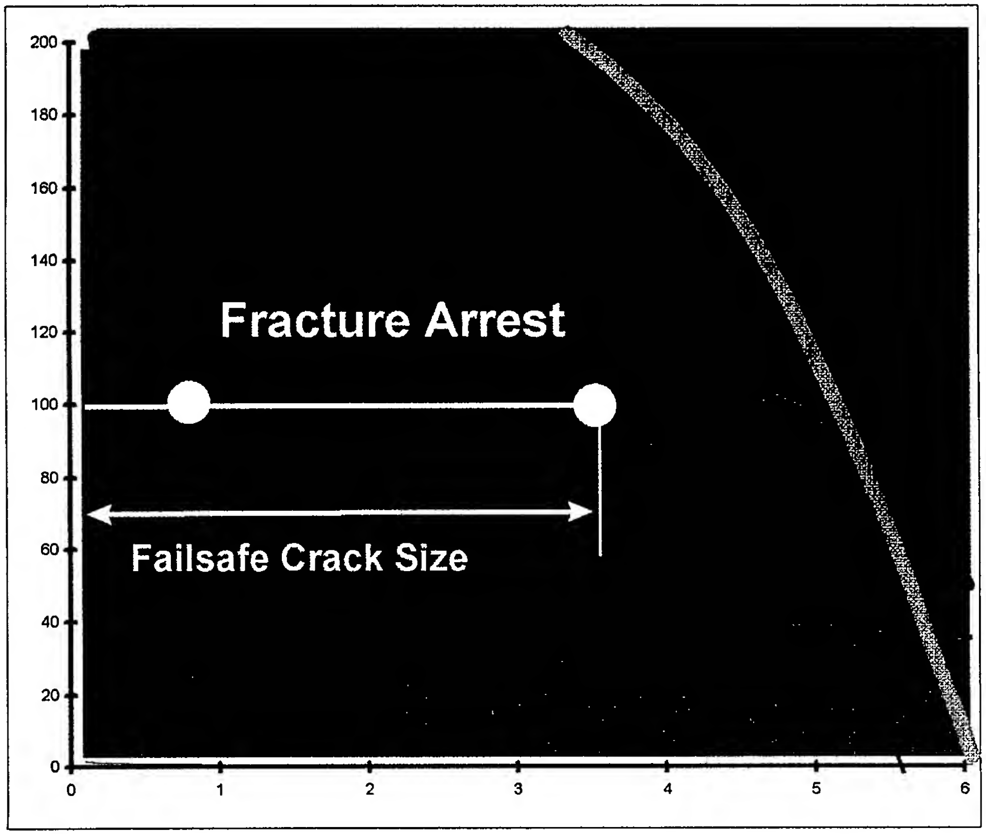


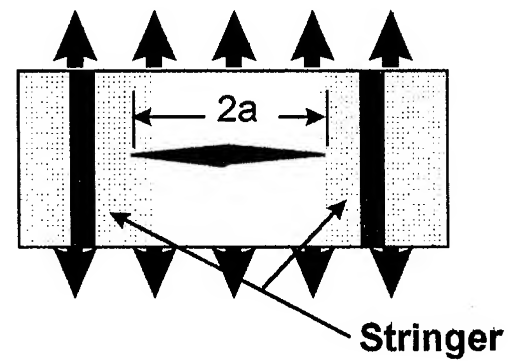
- Capable of Being Applied On Large Scale
- Possible to NDI Original Structure after Repair



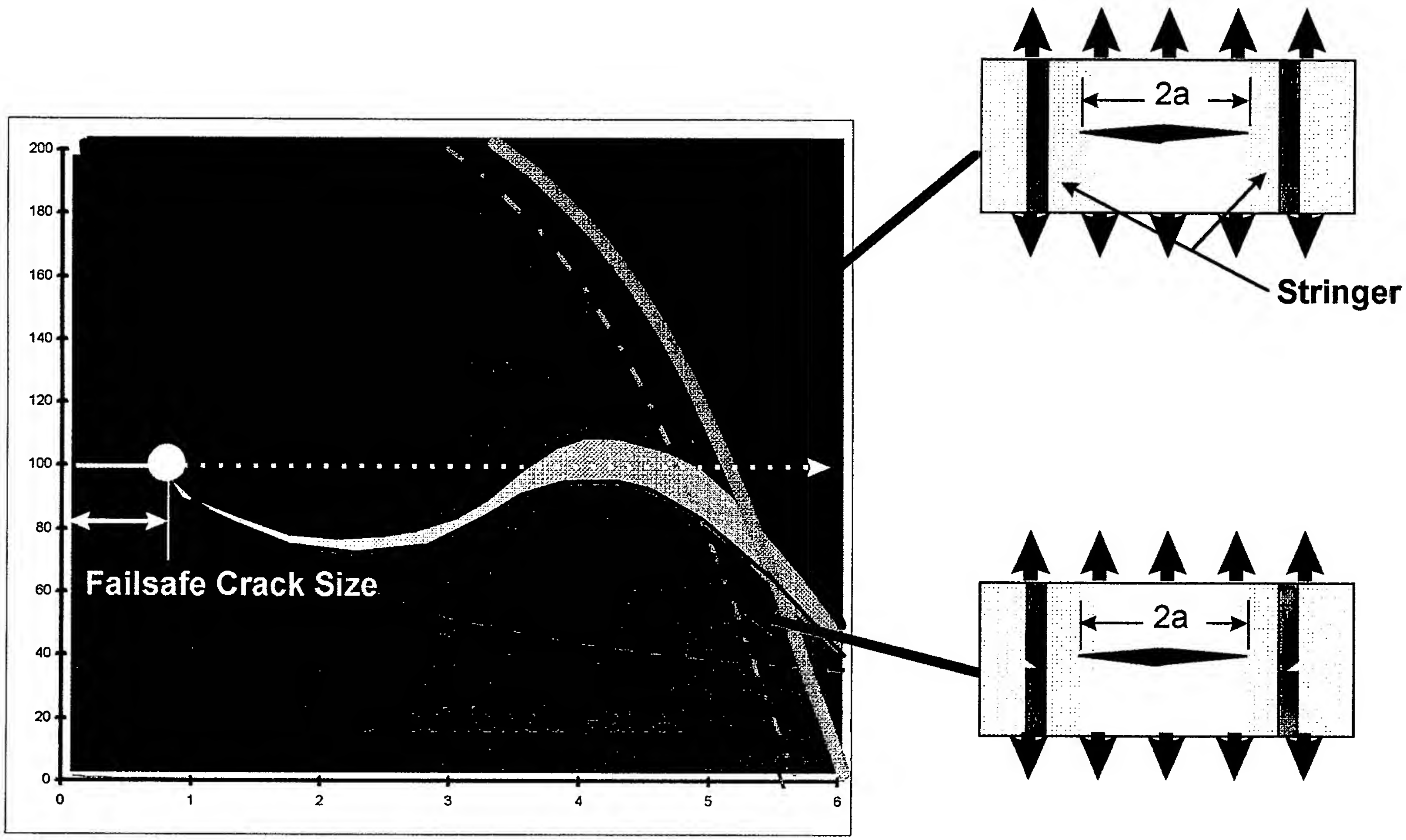
- Potential To Be Considered A Modification to Solve Problem
  - Not Just Treat a Symptom

Residual Strength Curves: Stiffened panel





Residual Strength With Damaged Stringer



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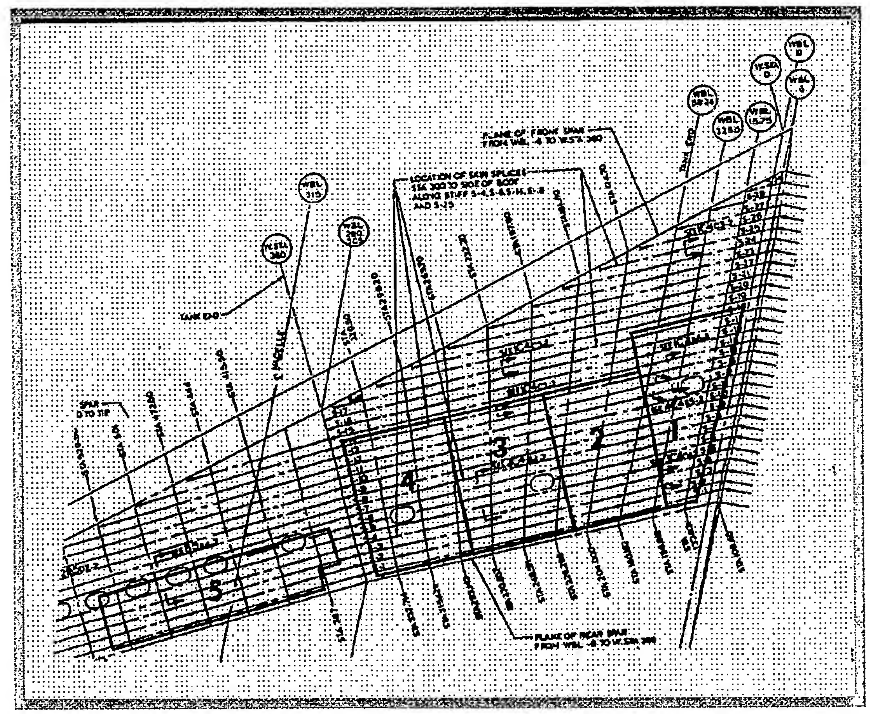
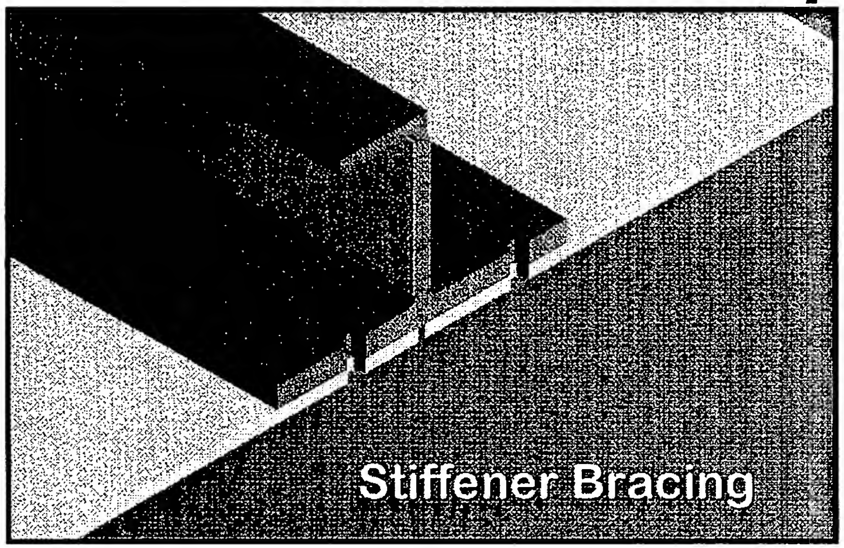
Current Problem For Joint- Stars Program

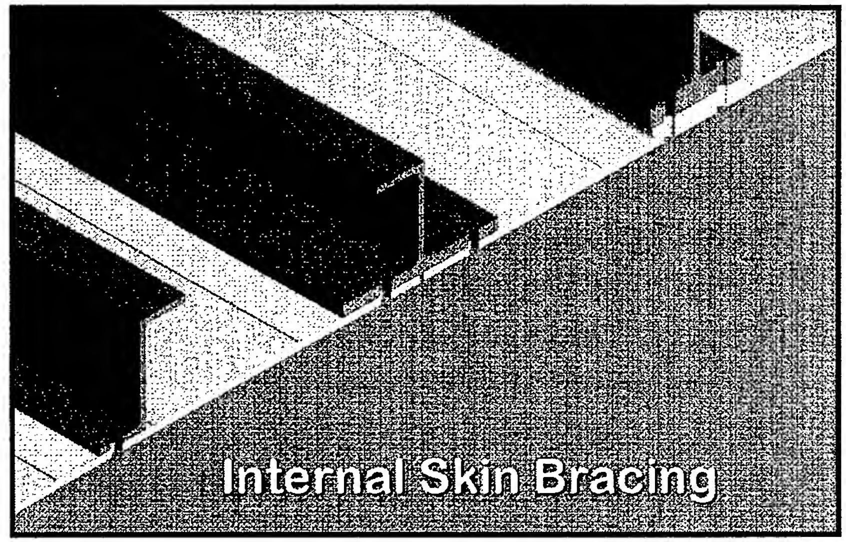
- Old B-707 Airframes
- Teardown Inspection Results
  - 6100 Holes Inspected
  - 1548 Had Cracks
- Probability of Failure
  - Becomes Unacceptable During Intended Life in the New Role

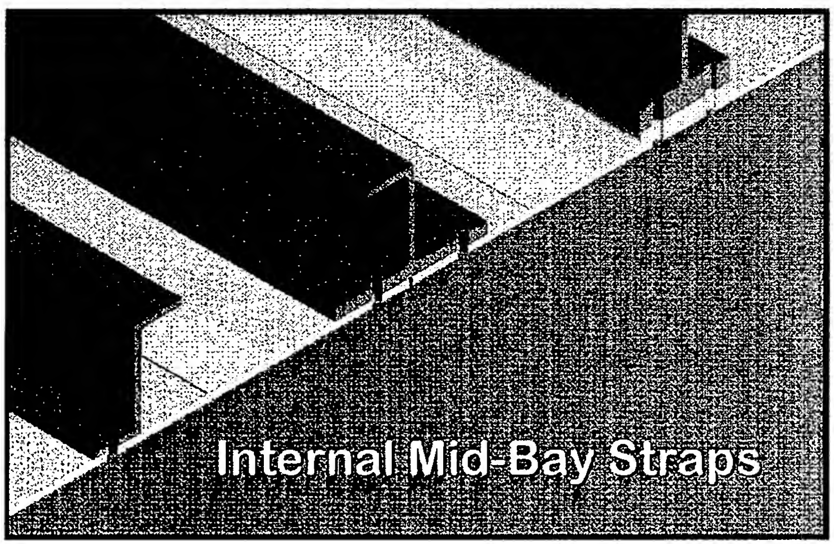
- Methods Being Considered
  - Detailed NDI
  - Cold Expansion of Holes
  - Component Replacement
  - Modification

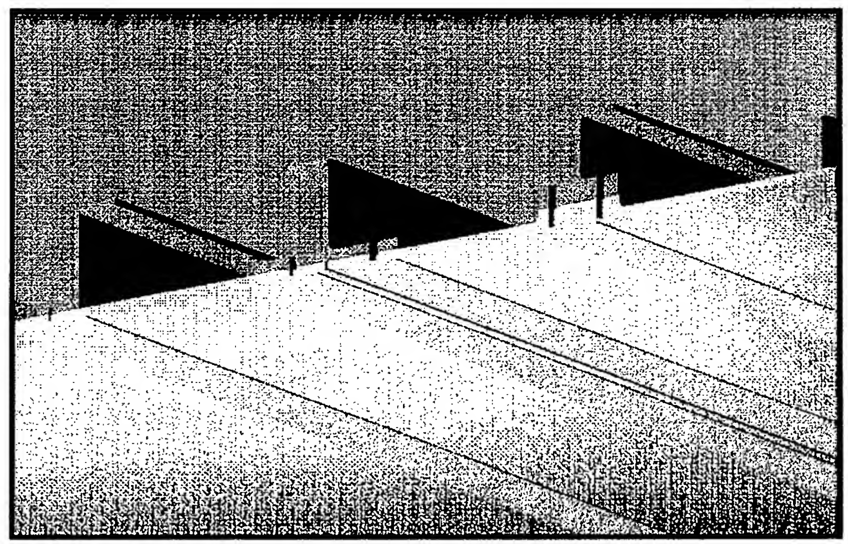


Bonded Repair Concepts









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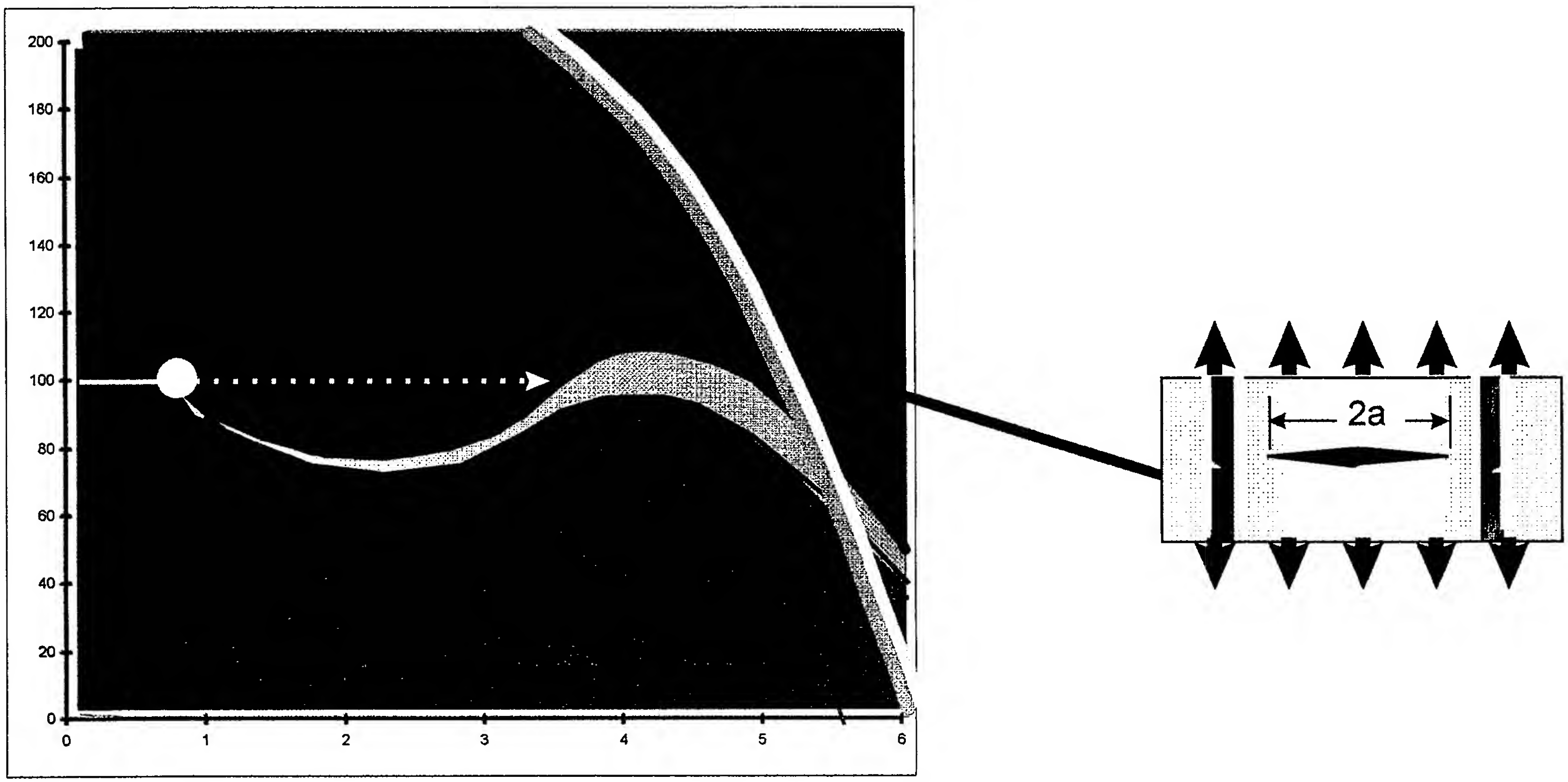
Restore Residual Strength Characteristics

- Restore Stringer Properties by Reinforcement
  - Regain Original Failsafety or Potentially Better Performance



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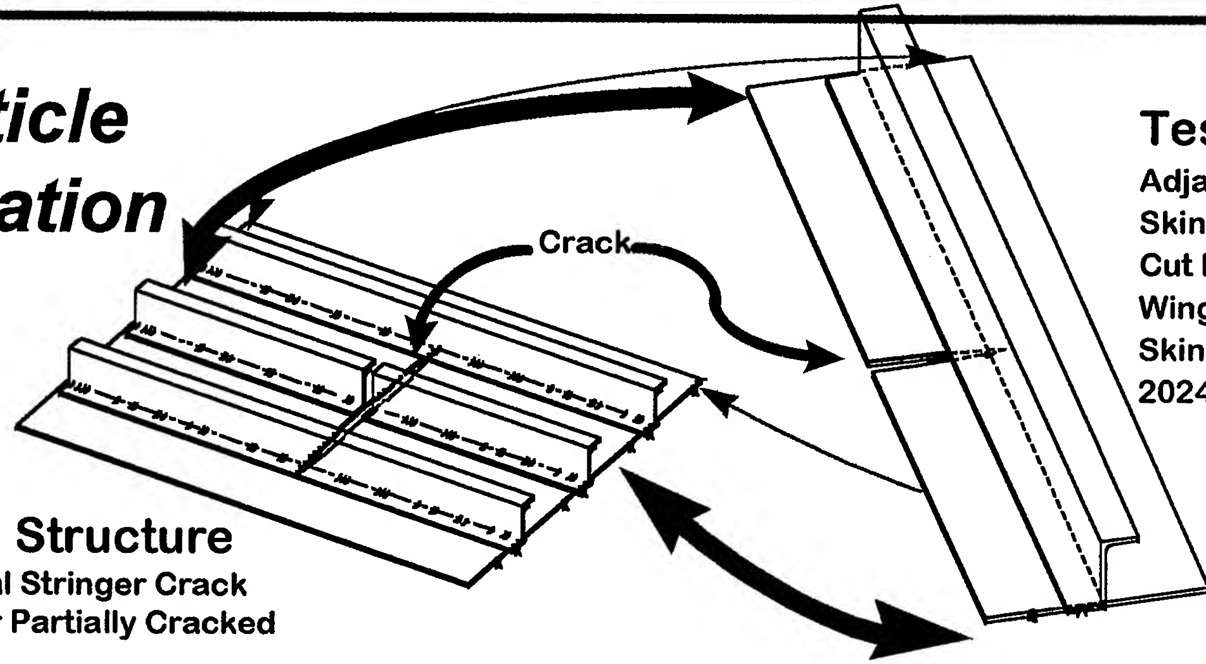
ARTI — Advanced Repair Technology International

Test Article Configuration

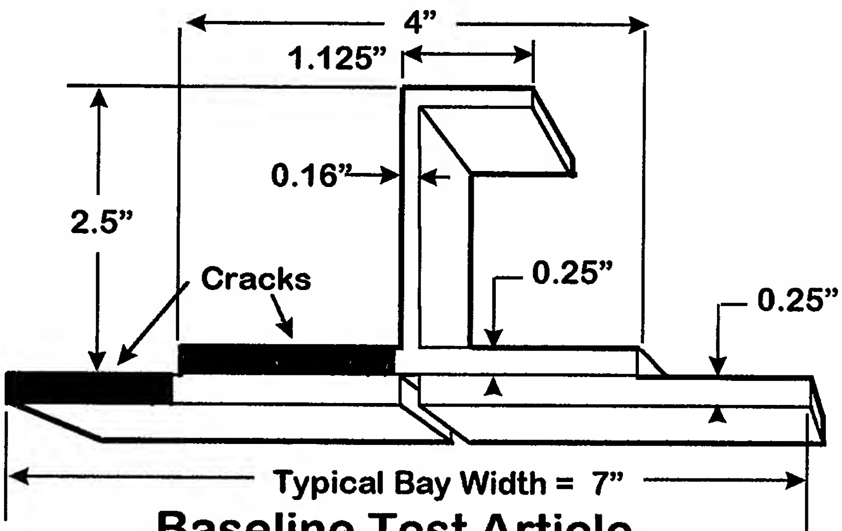


Simulated Structure

Two Bay Central Stringer Crack
Adjacent Stringer Partially Cracked

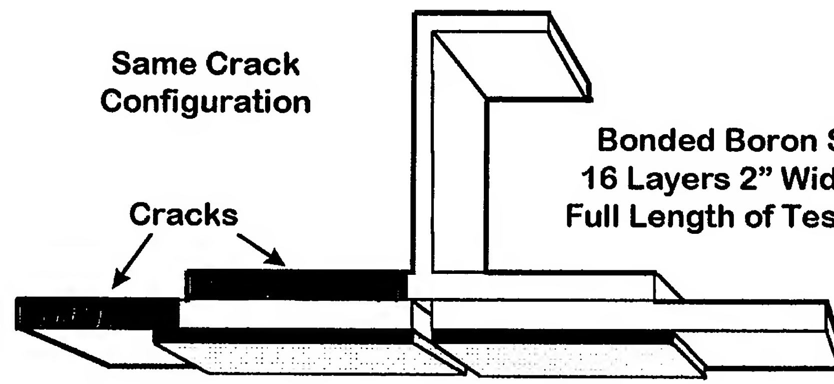
Test Article

Adjacent Stringer
Skin- One Bay Wide
Cut From B-727 Lower Wing Skin
Skin & Stringer Material 2024



Baseline Test Article

Section Through Cracked Zone

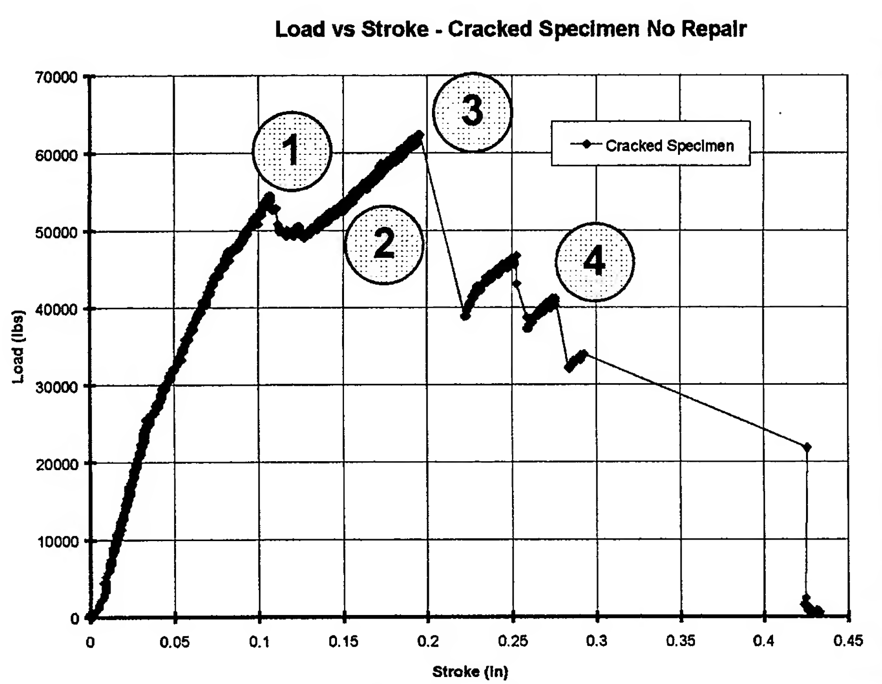


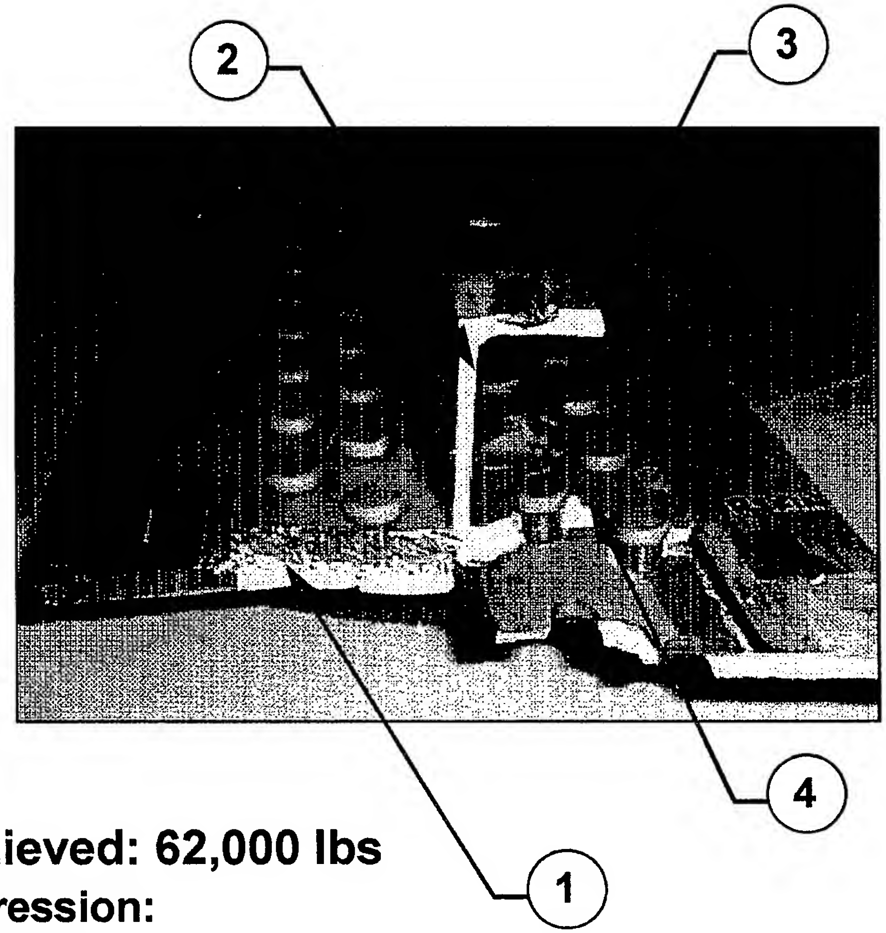
Repaired Test Article

Section Through Cracked Zone

The Bonded Repair Company

Test Results - Cracked Specimen





- **Max Load Achieved: 62,000 lbs**
  - **Crack Progression:**
    - » **Skin at Crack Location**
    - » **Stringer Web (Slow Growth)**
    - » **Stringer Flange (Sudden)**
    - » **Skin on Opposite Side Sudden)**

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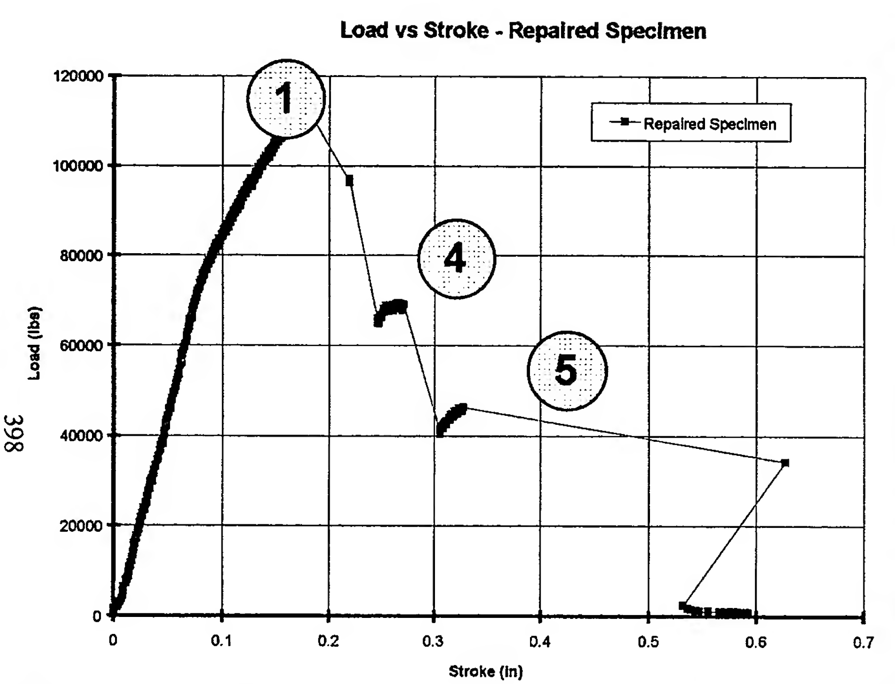
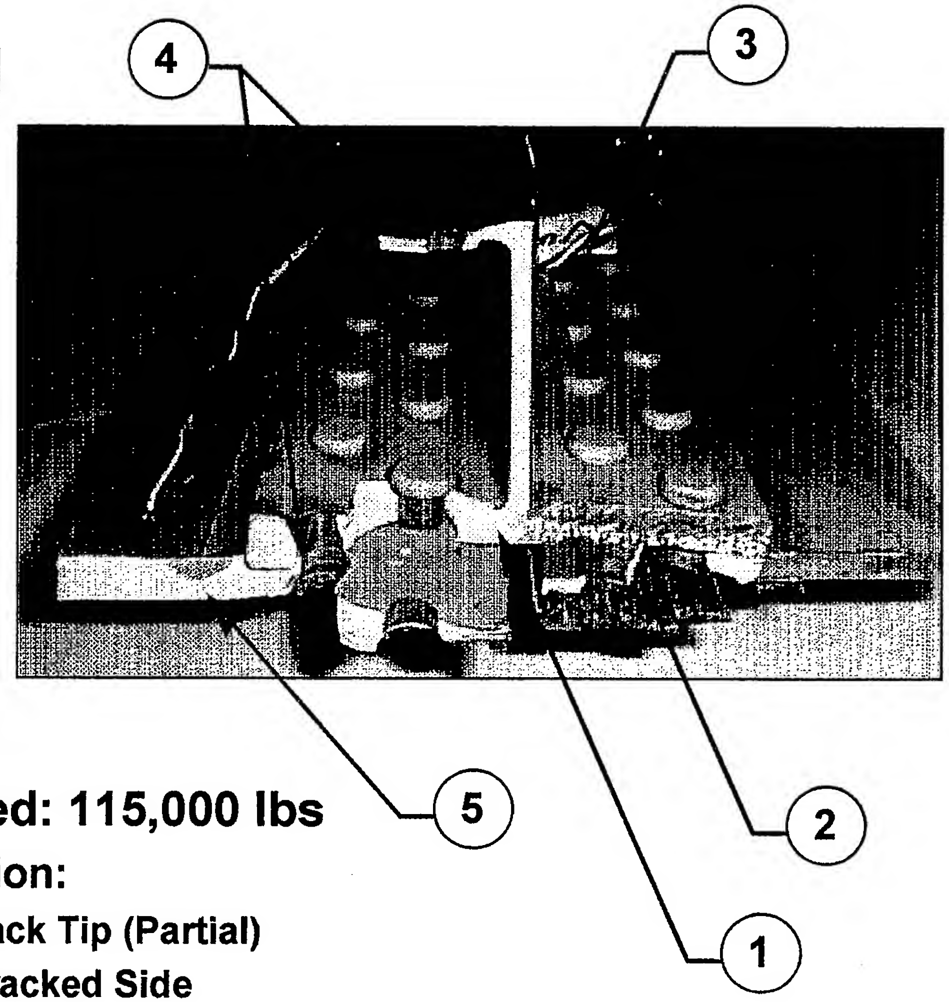
Test Results - Repaired Specimen



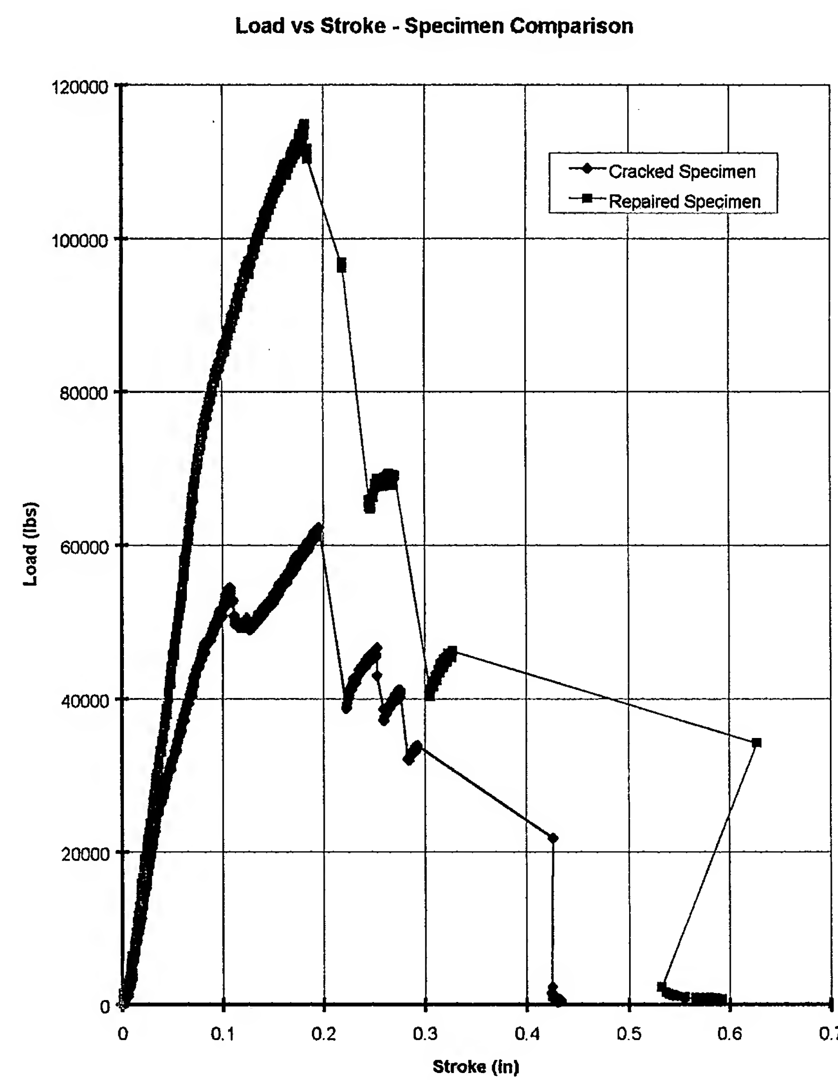


- **Max Load Achieved: 115,000 lbs**
  - **Crack Progression:**
    - » **Stringer at Crack Tip (Partial)**
    - » **Skin on the Cracked Side**
    - » **Stringer Web**
    - » **Flange and Skin on Opposite Side (Sudden)**

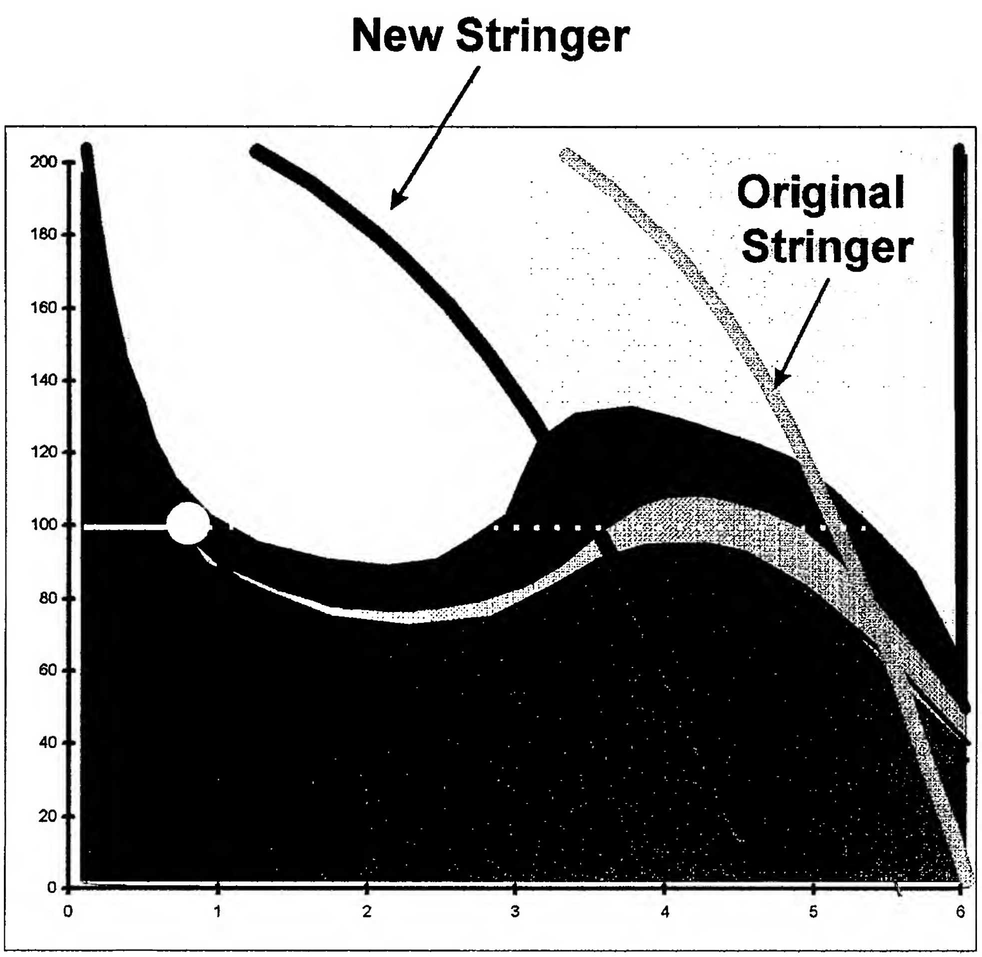
The Bonded Repair Company

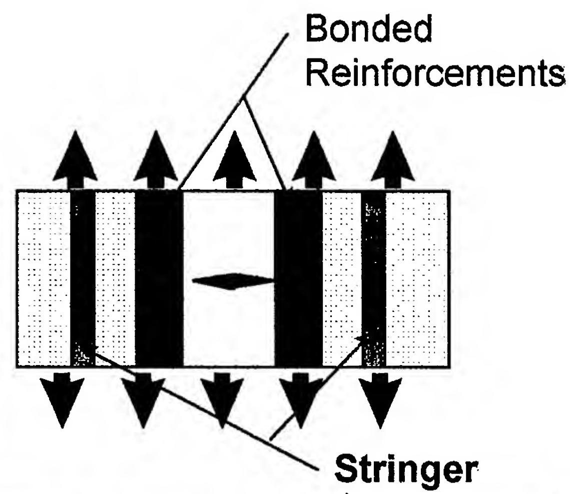
Test Results - Comparison

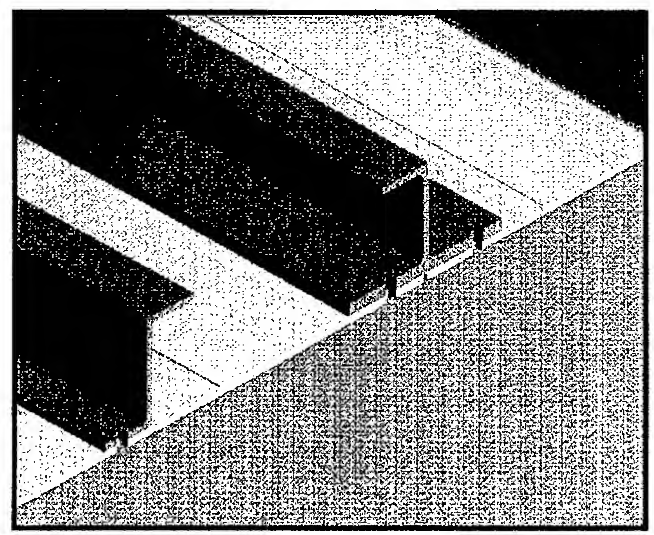
- Significantly Higher Load Carrying Capability in Repaired Configuration
- Change of Failure Mechanism
  - Lower Skin/Stringer Crack Opening Forces
  - Delay of Skin Failure That Leads to Stringer Failure
- Adhesive Successfully Transfers Skin Loads into Bonded Reinforcement



Modify Residual Strength Behavior



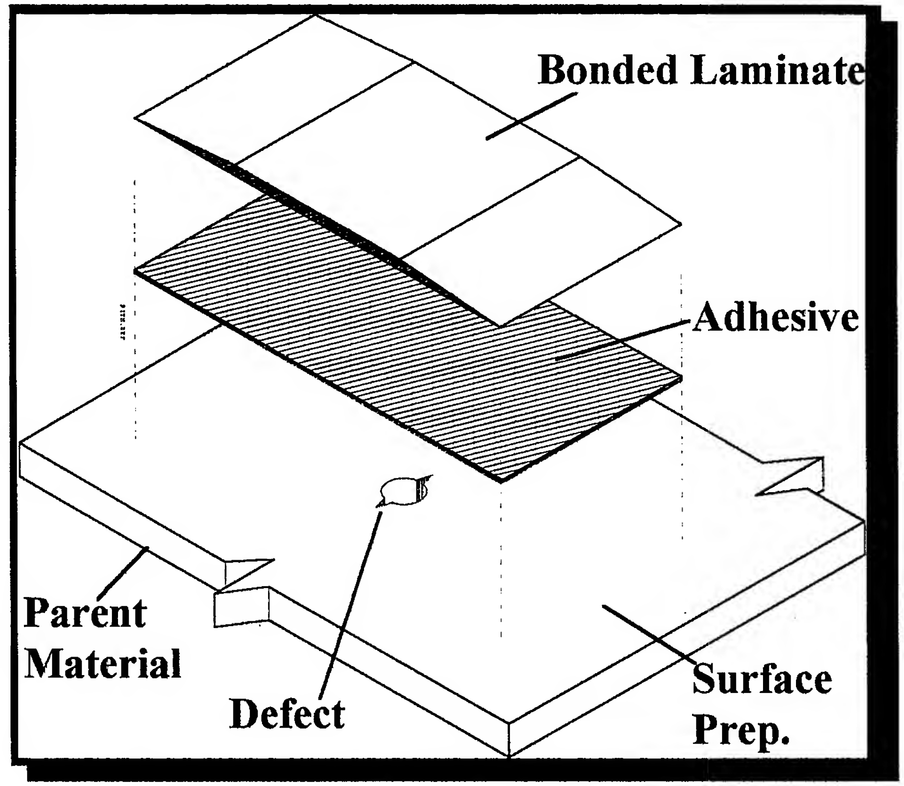




The Bonded Repair Company

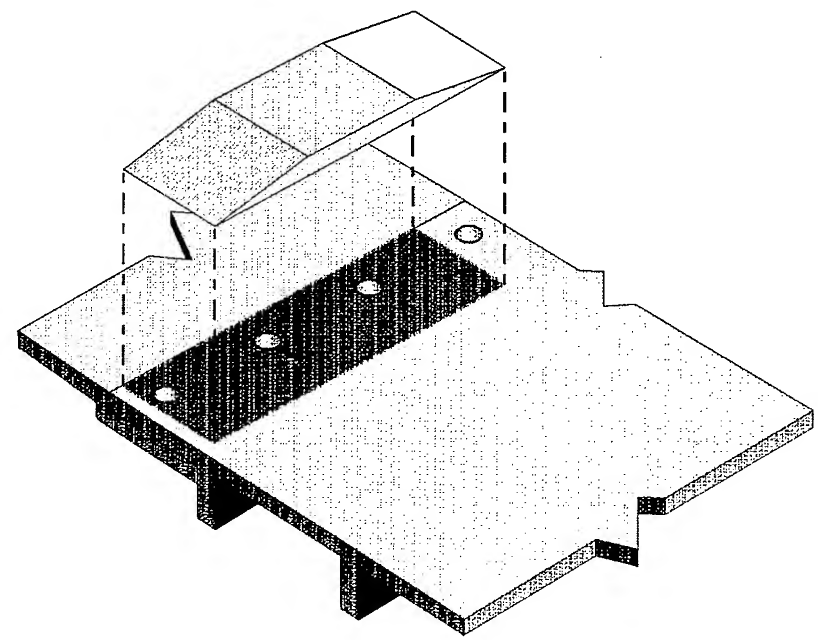
Bonded Repairs

- Are Just Another Resource
- Have Unique Properties Which Allow
  - Reduces stresses at driving the cracks
  - Provides a fully capable alternate load path
  - Installed with single side access
  - Suitable choice of reinforcing material will allow continued monitoring of the substructure.



Conclusion

- The Structural Integrity of Aging Aircraft Will Remain a Constant Safety Issue
- The Continued Existence Will Always Be A Balance Between Safety and Economics
- Innovative Use Of Bonded Repair Techniques Will Give a Greater Scope to Obtain That Desired Economic Solution



SESSION V

NONDESTRUCTIVE EVALUATION/ INSPECTION

Chairman - *M. Paulk*
Air Force NDI Program Office

QUANTITATIVE ASSESSMENT OF THE DETECTABILITY OF CERAMIC INCLUSIONS IN STRUCTURAL TITANIUM CASTINGS BY X-RADIOGRAPHY

Fred R. Child
Ward. D. Rummel
David. H. Phillips
Leonard W. Liese

D&W Enterprises, LTD.

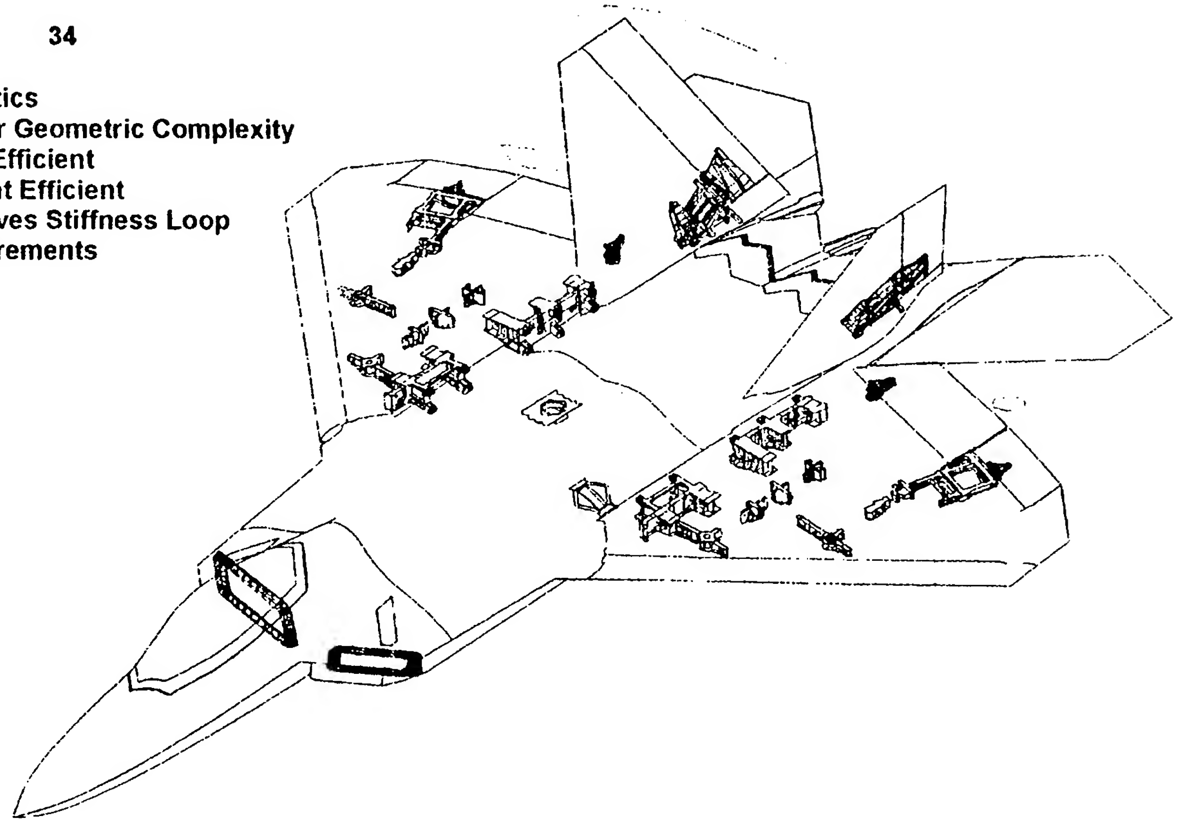
BACKGROUND

- Ti-Hip Castings employed for fracture / critical components
- Traditional Forging applications
- Wing-attach fittings, Control surface actuators
- Typical defects - Shell Inclusions - Moderately acceptable in aerospace stiffness applications
- Shell Inclusions - extremely critical to damage tolerant structure
- Detectability must be QUANTIFIED & incorporated into life cycle management

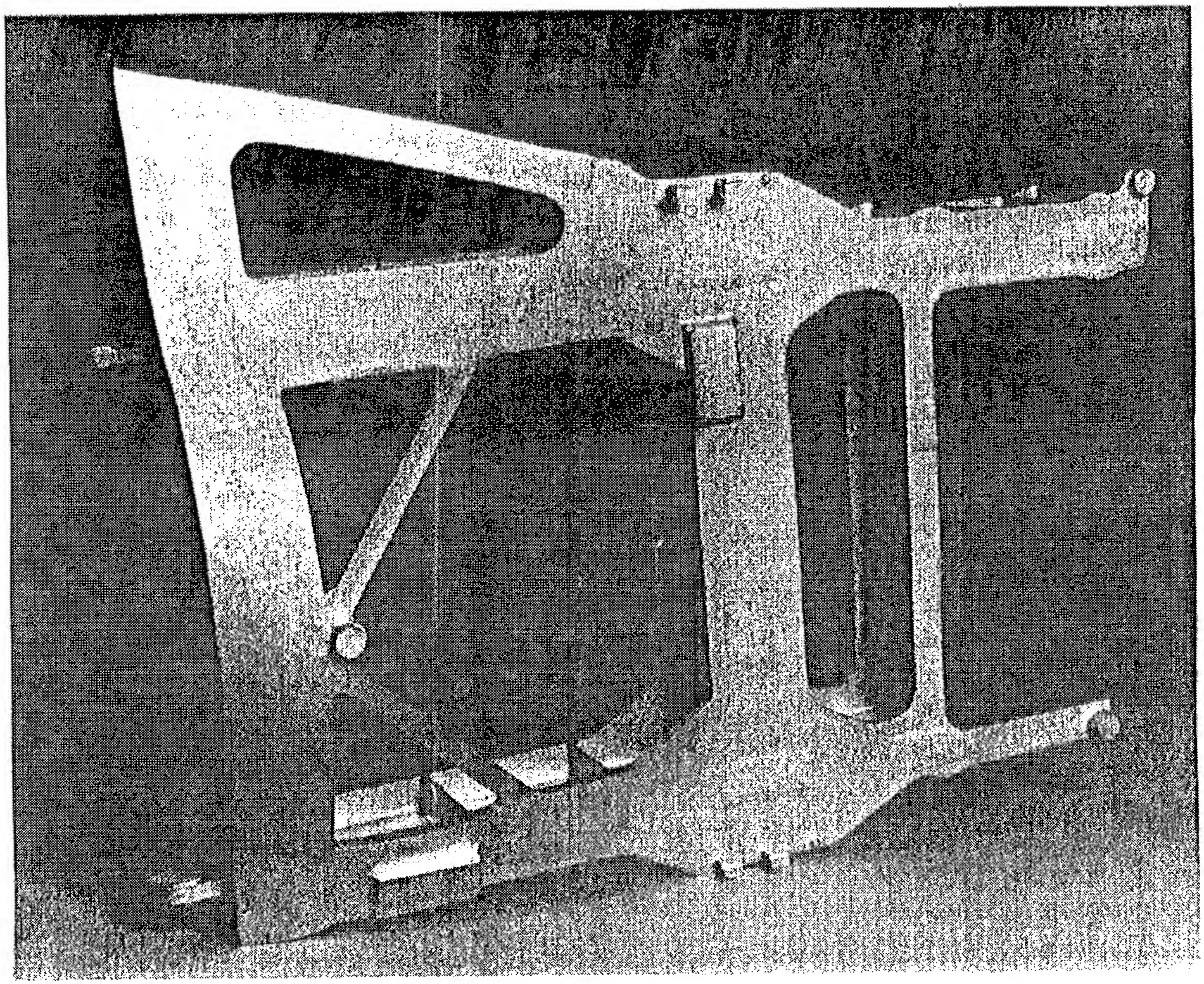
Ti HIP CASTING APPLICATIONS

- Total No. 34

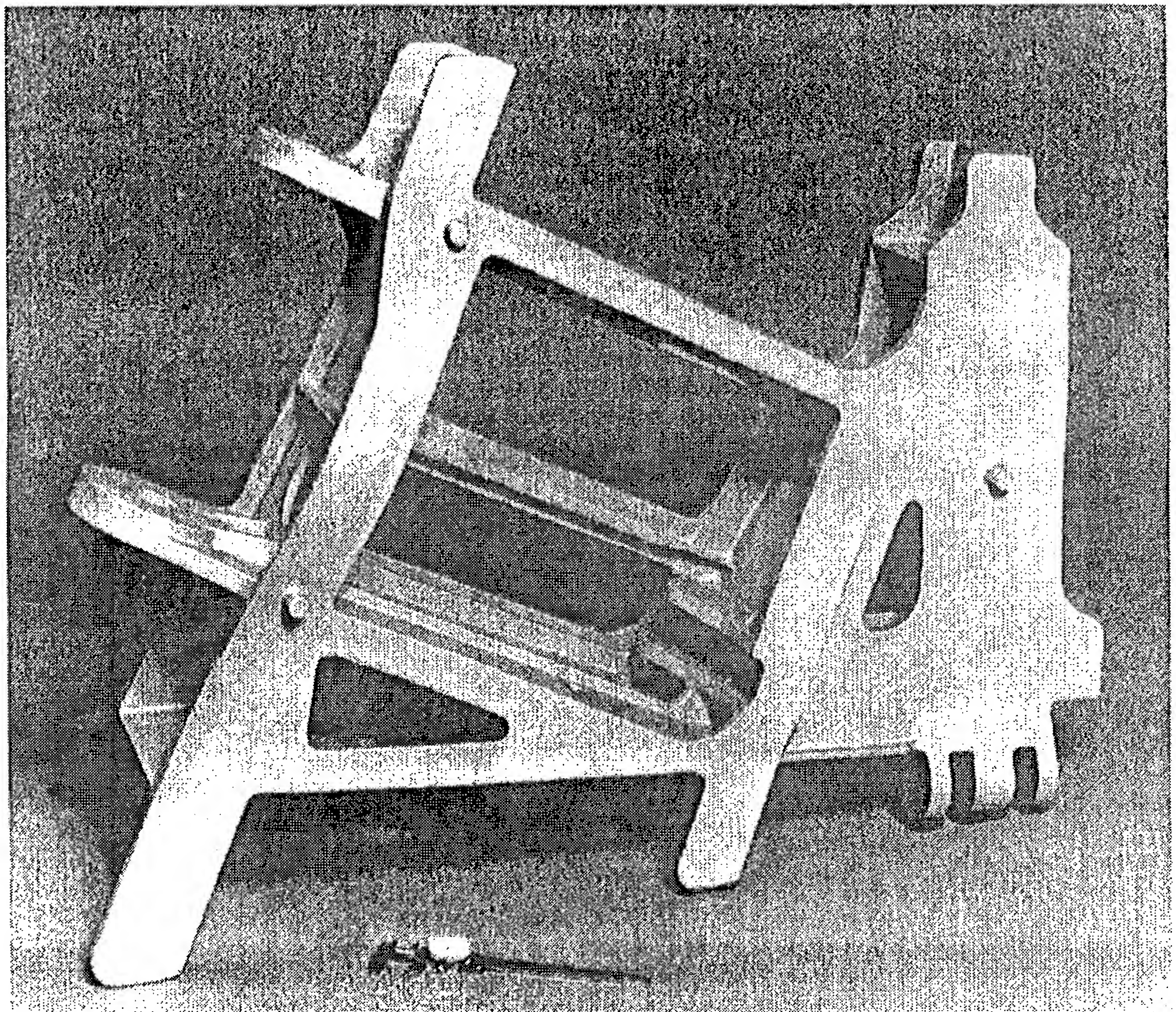
- Characteristics
  - Higher Geometric Complexity
  - Cost Efficient
  - Weight Efficient
  - Achieves Stiffness Loop Requirements

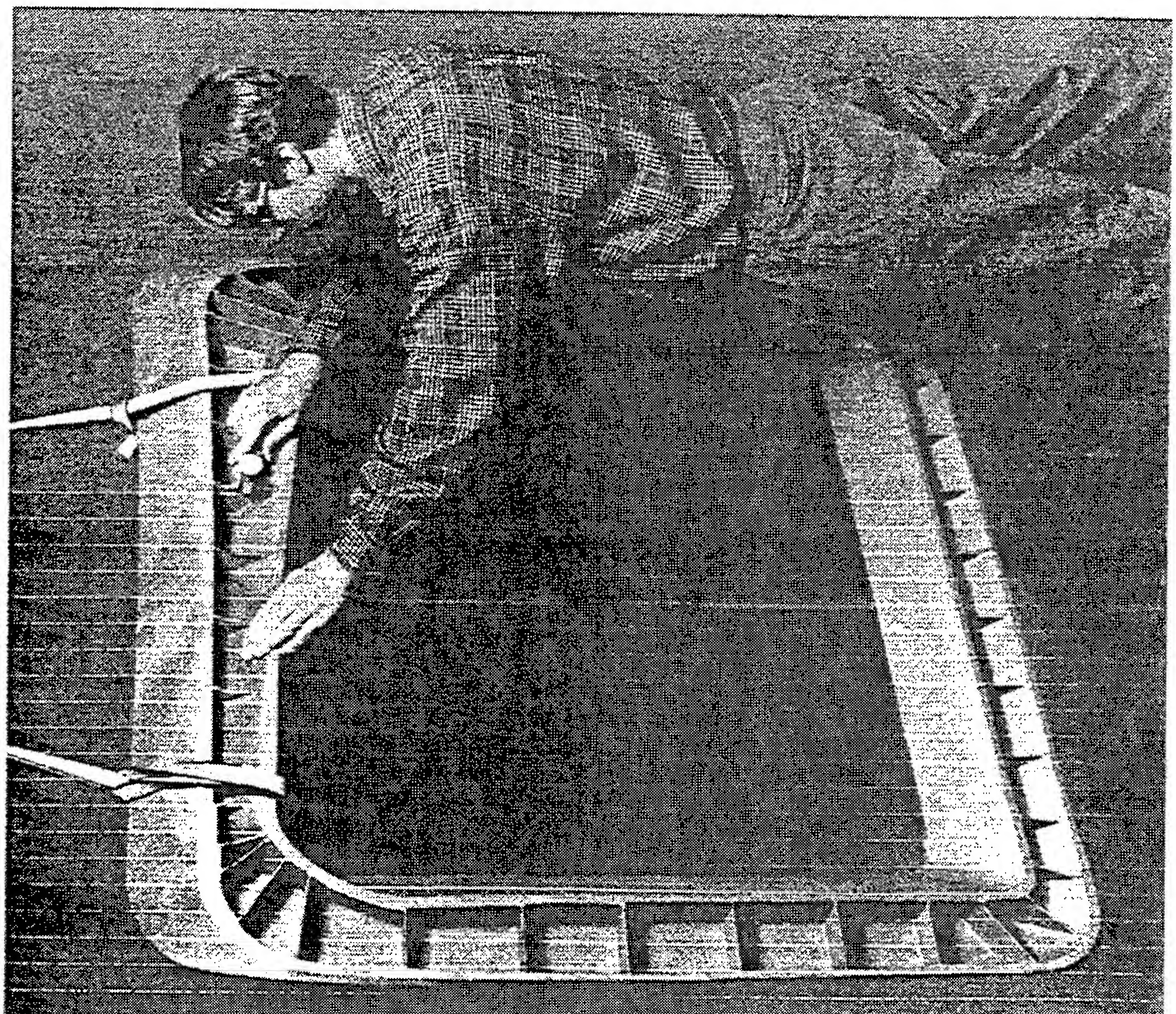


AILERON WAYBACK



RUDDER ACTUATOR HOUSING





OBJECTIVES

- Quantify shell detectability by X-radiography
- Modify process to enhance inspectability
- Incorporate shell detectability in design and life cycle management

APPROACH

- Obtain facecoat material from several vendors
- Select / size shell samples in increasing diameters
- X-radiograph varying diameter shell on varying Ti thickness
- Utilize vendor peculiar exposure techniques
- Plot POD vs Shell Diameter curves for each facecoat & Ti thickness

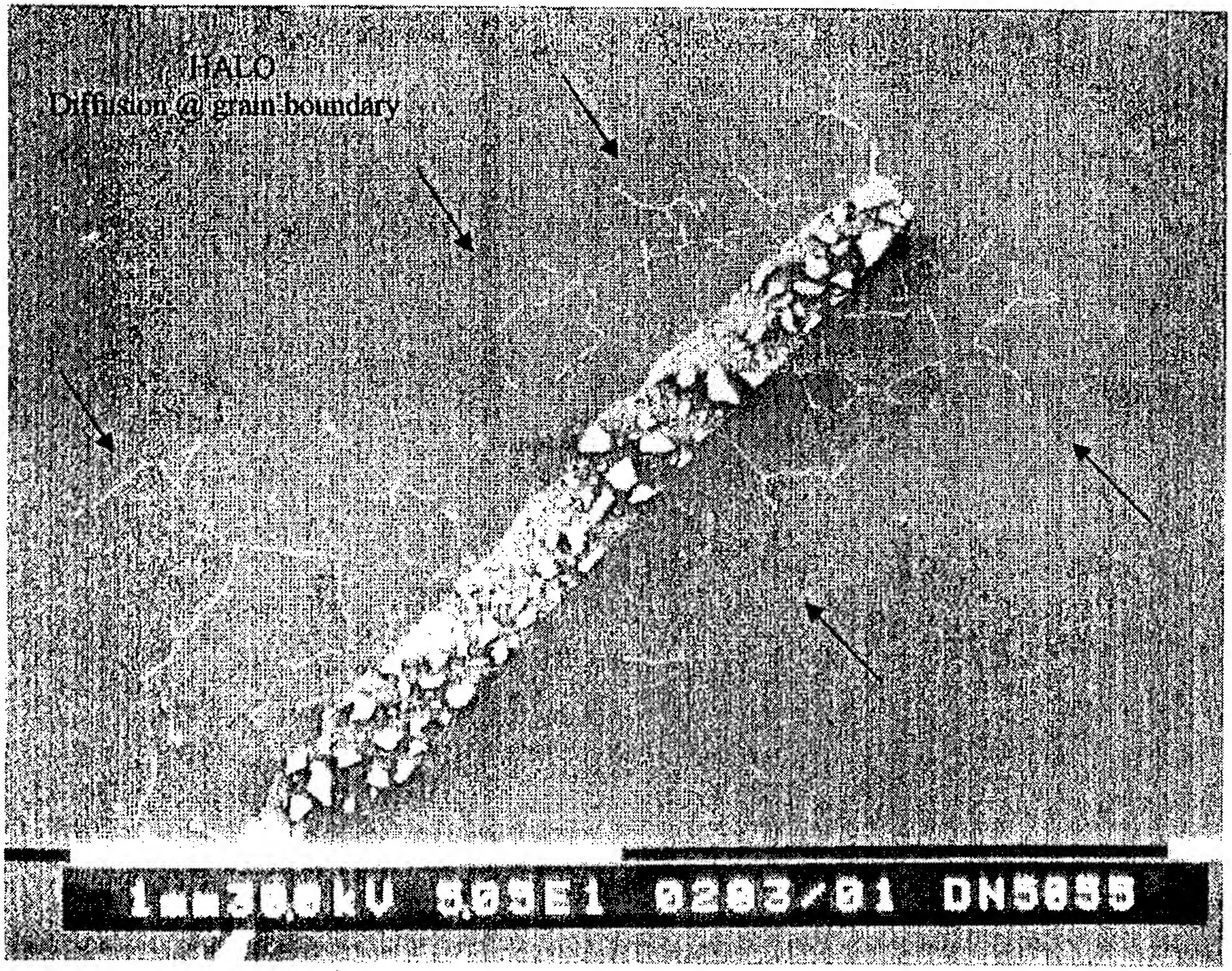
APPROACH (cont.)

- Plot Shell Diameter @ 90% POD vs Ti Thickness for each material
- Use multiple shell thickness (stacks) for non-planar orientation
- Determine effective size of the inclusion & surrounding "halo"
- Validate results with seeded "cast-in defect" samples
- Explore different facecoat material to enhance detectability
- Modify designs to accommodate reliability estimates from POD

DEFECT TERMINOLOGY

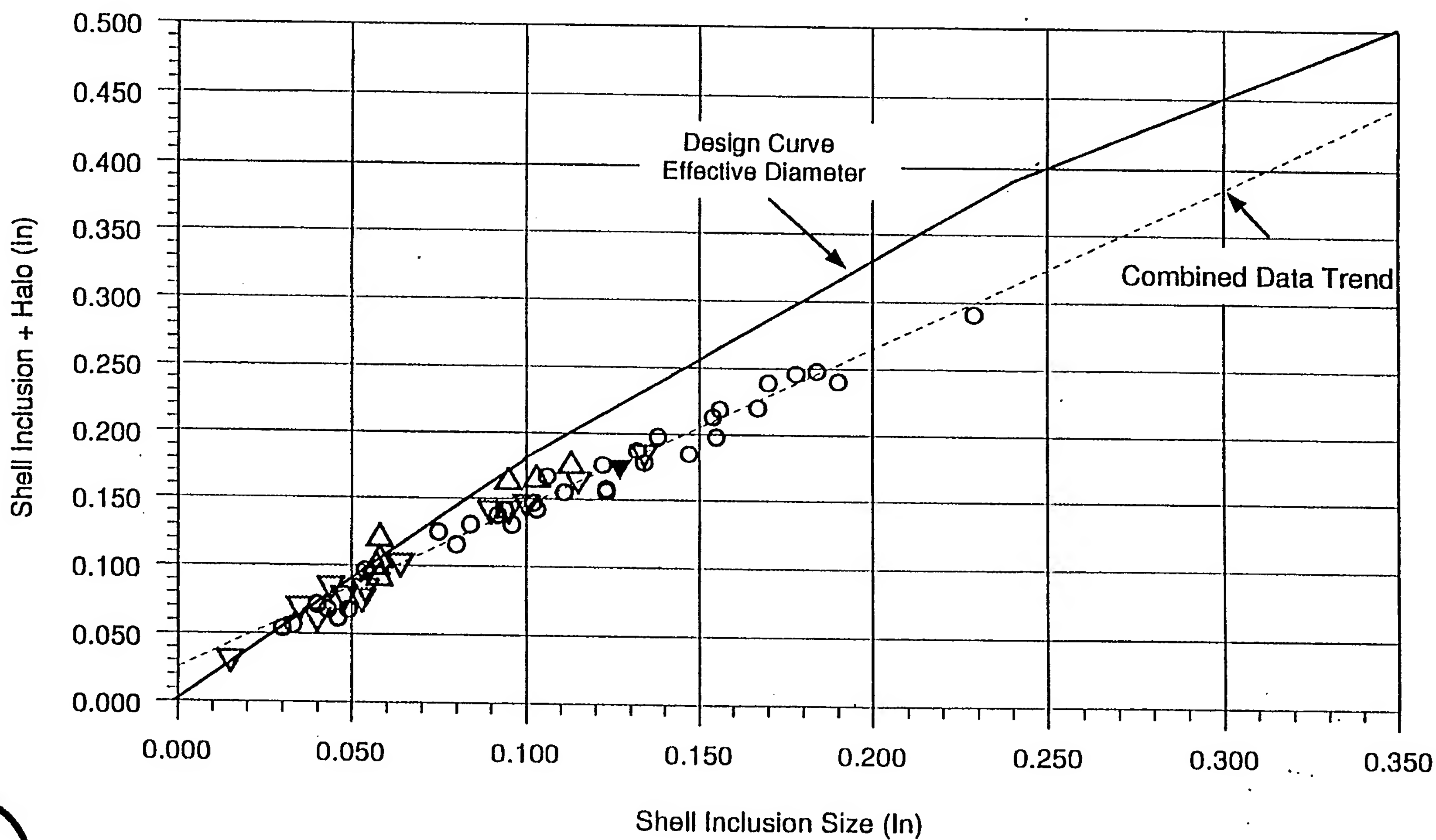
- SHELL - Ceramic inclusion resulting from spalling of casting mold facecoat material
- FACECOAT - Primary ceramic constituent of casting mold, usually applied to first 3 - 5 layers of the mold adjacent to wax pattern
- BACKUP STUCCO - Binding material, usually silica based flour, applied to secondary layers of facecoat to provide strength and thickness to the mold
- HALO - Media surrounding facecoat inclusion, generally consisting of dissolved facecoat consituents and Ti with high alpha case content

TYPICAL SHELL INCLUSION



EFFECTIVE DEFECT SIZE

Shell Inclusion Size vs Total Defect Size



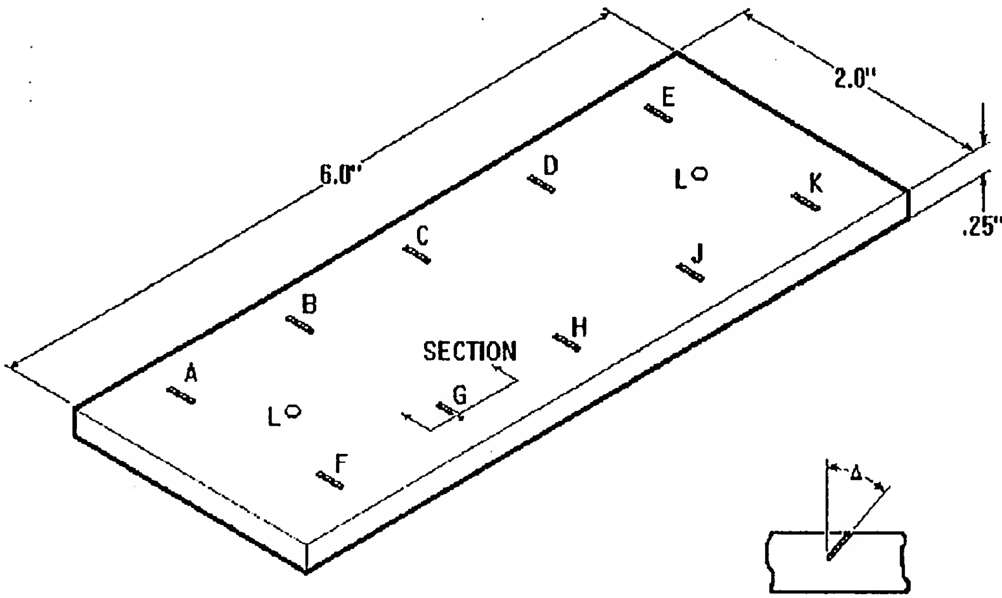
EXPERIMENT PARAMETERS

- **4 Different Vendors**
- **10 Facecoat Chemistries**
- **8 Inclusion Sizes - (0.030” - 0.250”)**
- **11 Different Ti Thicknesses (0.375” - 3.25”)**
- **7 Different Film Readers**
- **Planar Orientation (0.0035” - 0.0075”)**
- **Multiple Thickness Inclusions**
- **5 Angular Oriented Samples (15^0, 30^0, 45^0, 60^0, &90^0)**

IMAGE ANALYSIS

- Shell Detection - The pattern recognition of a subtle change in density (Signal / Noise) produced by the inclusion compared to that of the surrounding medium.
- Signal to Noise / Contrast
  - Varies exponentially with increasing thickness
  - Minimal at shell thickness below 0.005" vs Ti thickness
  - Decreases with shell dissolution / reaction in Ti matrix
  - Typically 0.02 - 0.08 H&D for shell
- Pattern Recognition
  - Resultant of scatter or mottle, rather than contrast from typical absorption differences
  - 0.030" diameter is the lower limit for reliable pattern recognition

ORIENTATION STANDARD

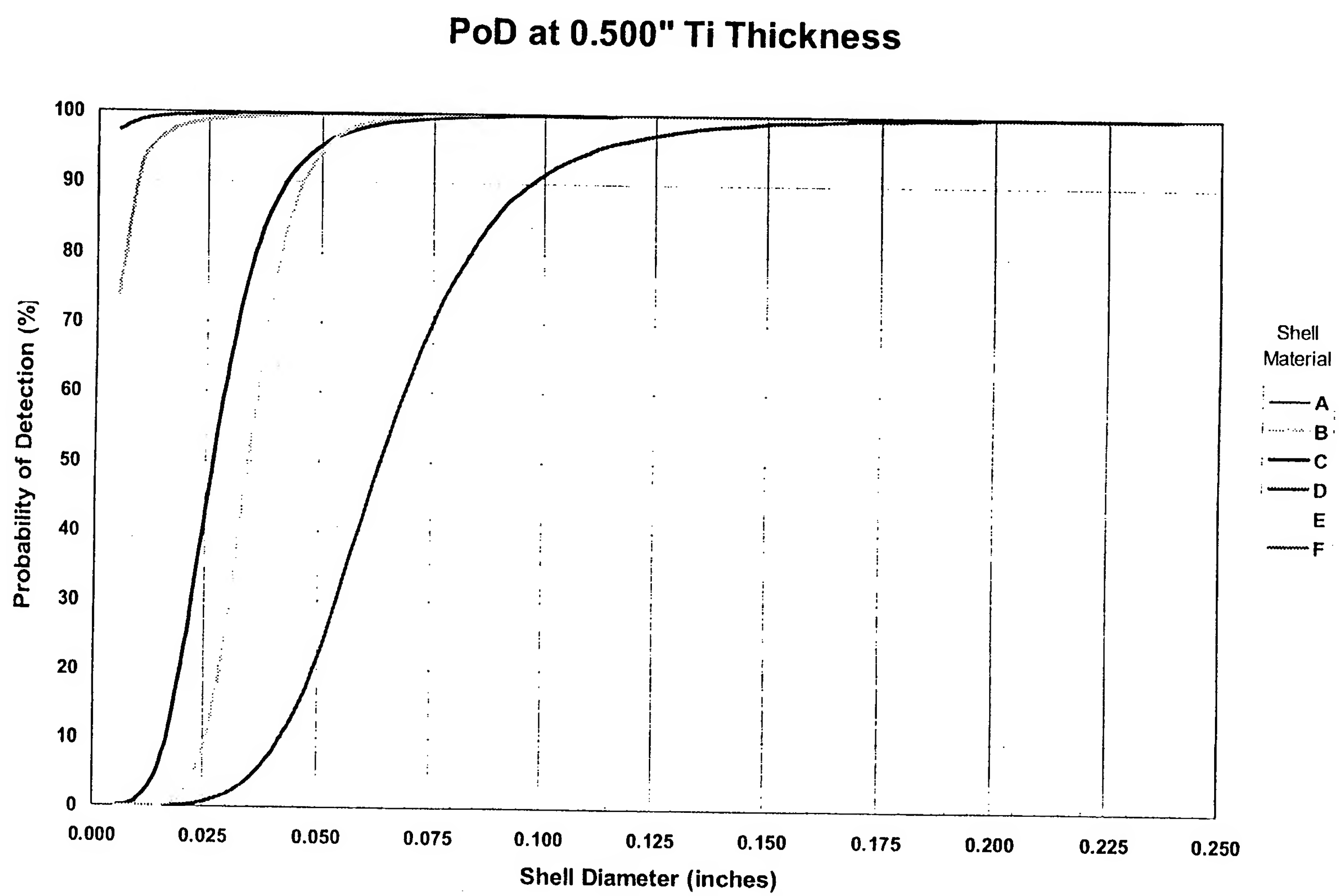


SECTION
(SLOTS G, H, J, & K)

ALL SLOTS ARE .005"-.007" WIDE

SLOT	DEPTH X LENGTH		
A	.010"	.090"	
B	.010"	.120"	
C	.010"	.150"	
D	.020"	.090"	
E	.020"	.120"	
F	.020"	.150"	
SLOT	**DEPTH X LENGTH X ANGLE**		
G	.020"	.090	Δ 15°
H	.020"	.090	Δ 30°
J	.020"	.090	Δ 45°
K	.020"	.090	Δ 60°
HOLE	**DEPTH X DIAMETER**		
L	.010"	.125"	

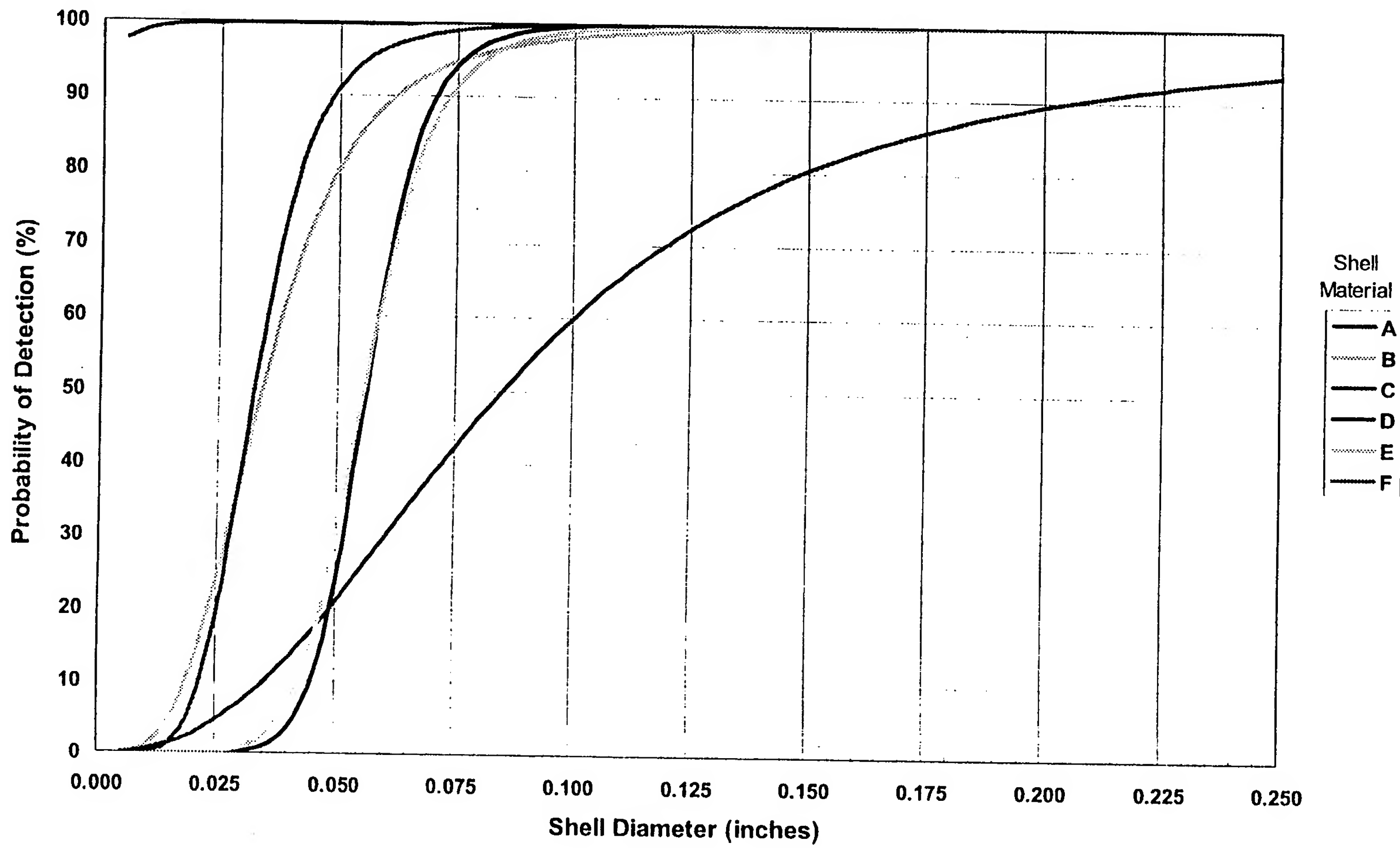
RESULTS



PoD at 0.500" Ti Thickness
Probability of Detection (%)
100
90
80
70
60
50
40
30
20
10
0
0.000
0.025
0.050
0.075
0.100
0.125
0.150
0.175
0.200
0.225
0.250
Shell Diameter (inches)
Shell Material
A
B
C
D
E
F

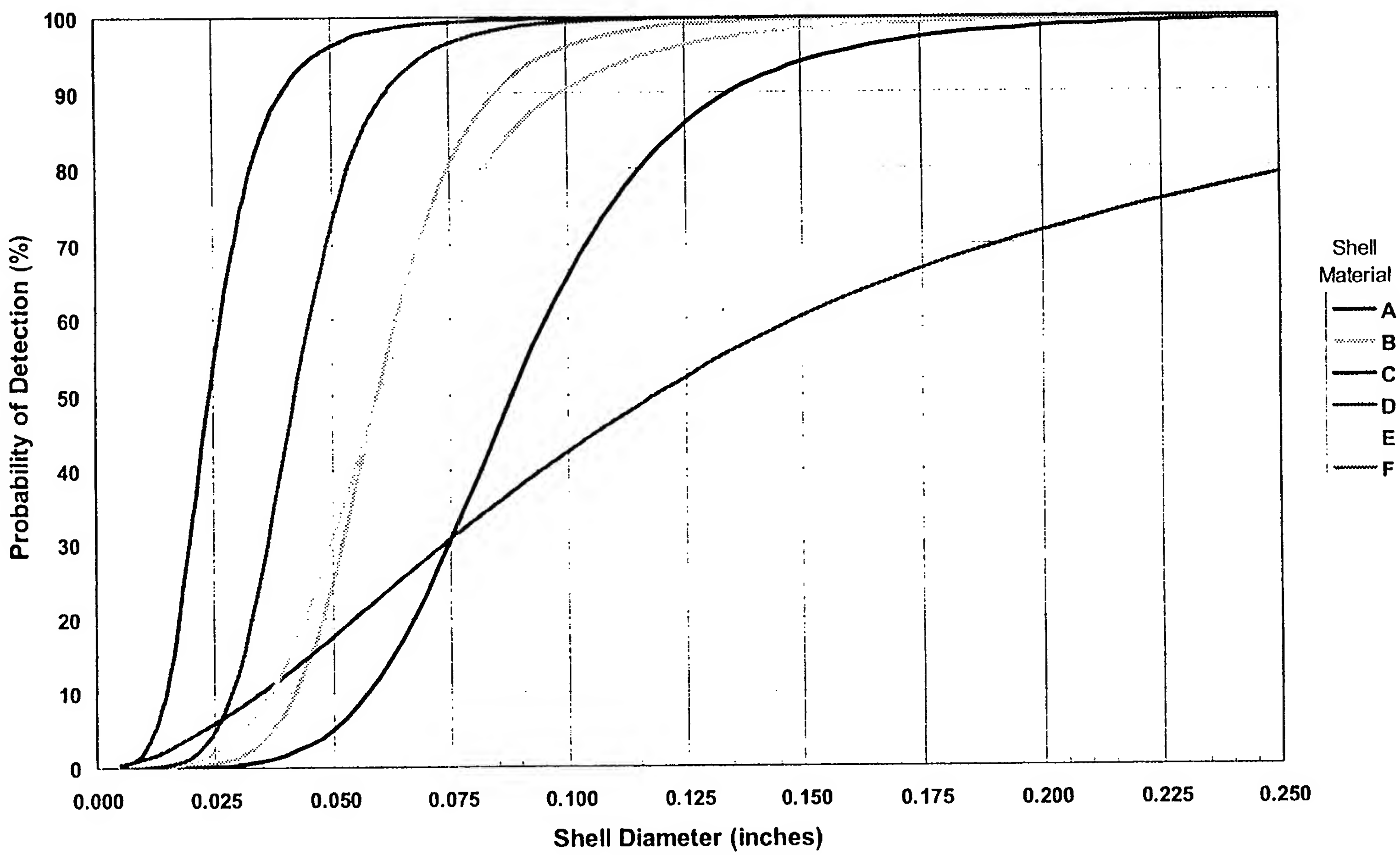
RESULTS

PoD at 0.704" Ti Thickness



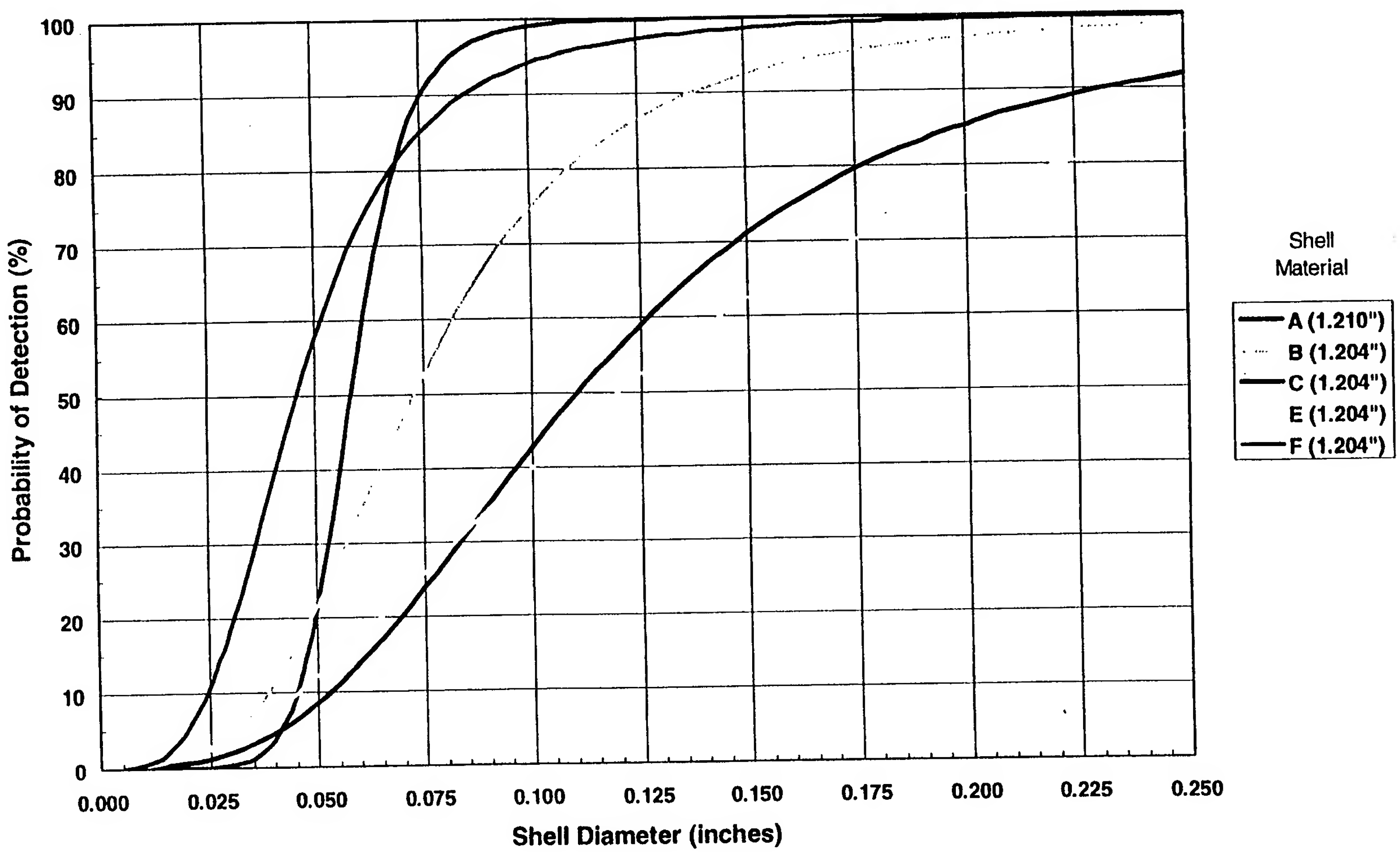
RESULTS

PoD at 0.904" Ti Thickness

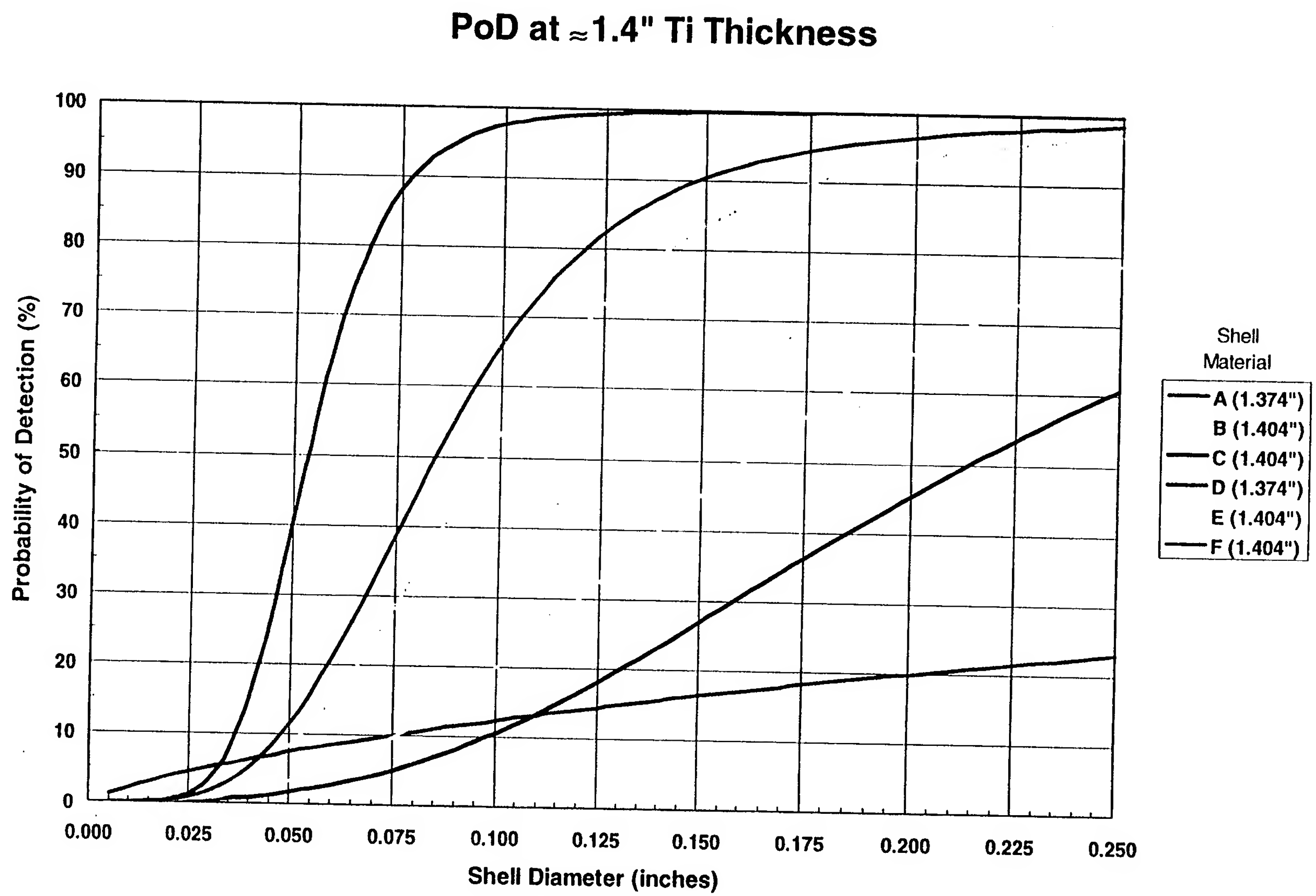


RESULTS

PoD at ≈1.2" Ti Thickness



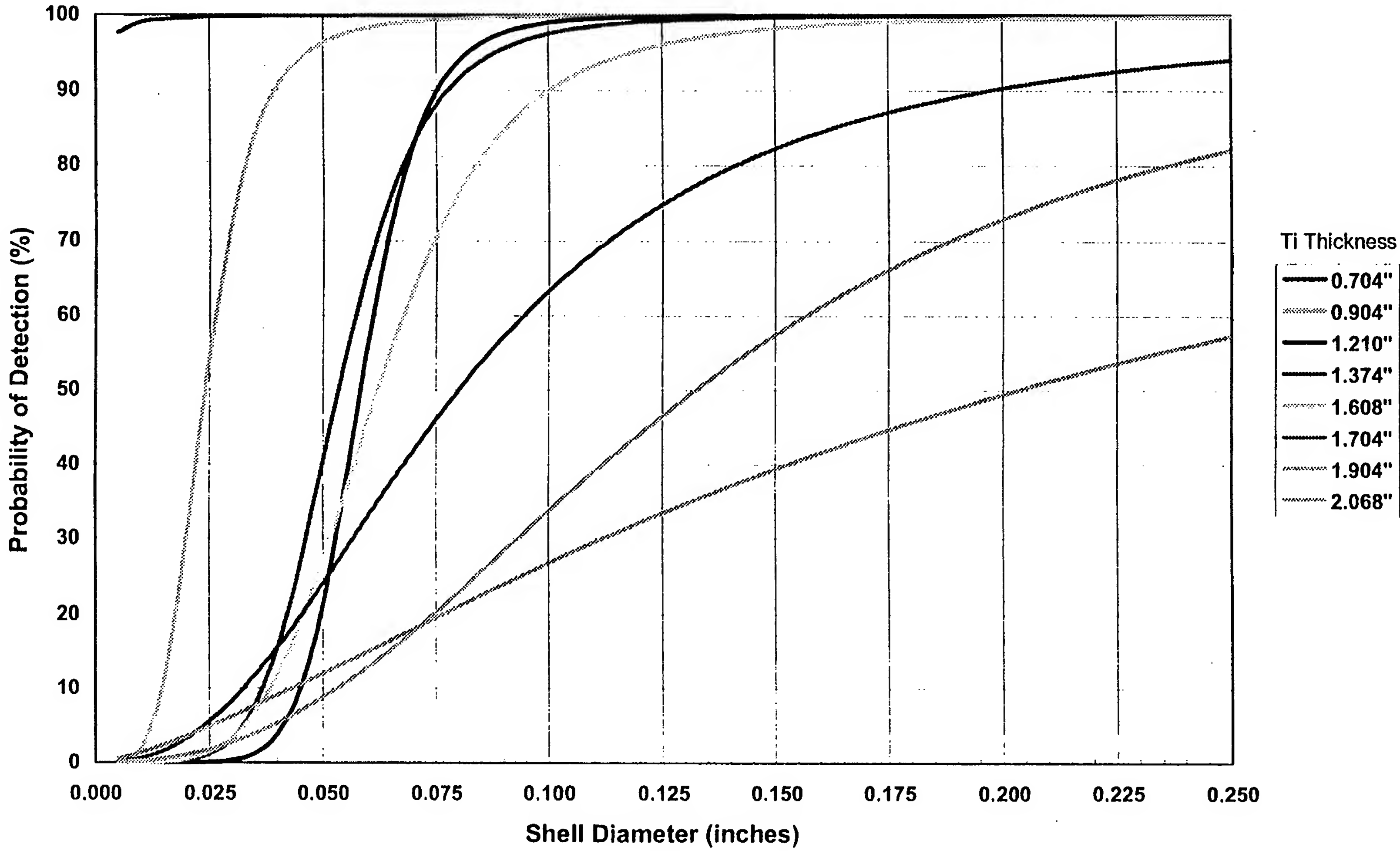
RESULTS



PoD at ≈1.4" Ti Thickness
Probability of Detection (%)
100
90
80
70
60
50
40
30
20
10
0
0.000
0.025
0.050
0.075
0.100
0.125
0.150
0.175
0.200
0.225
0.250
Shell Diameter (inches)
Shell Material
A (1.374")
B (1.404")
C (1.404")
D (1.374")
E (1.404")
F (1.404")

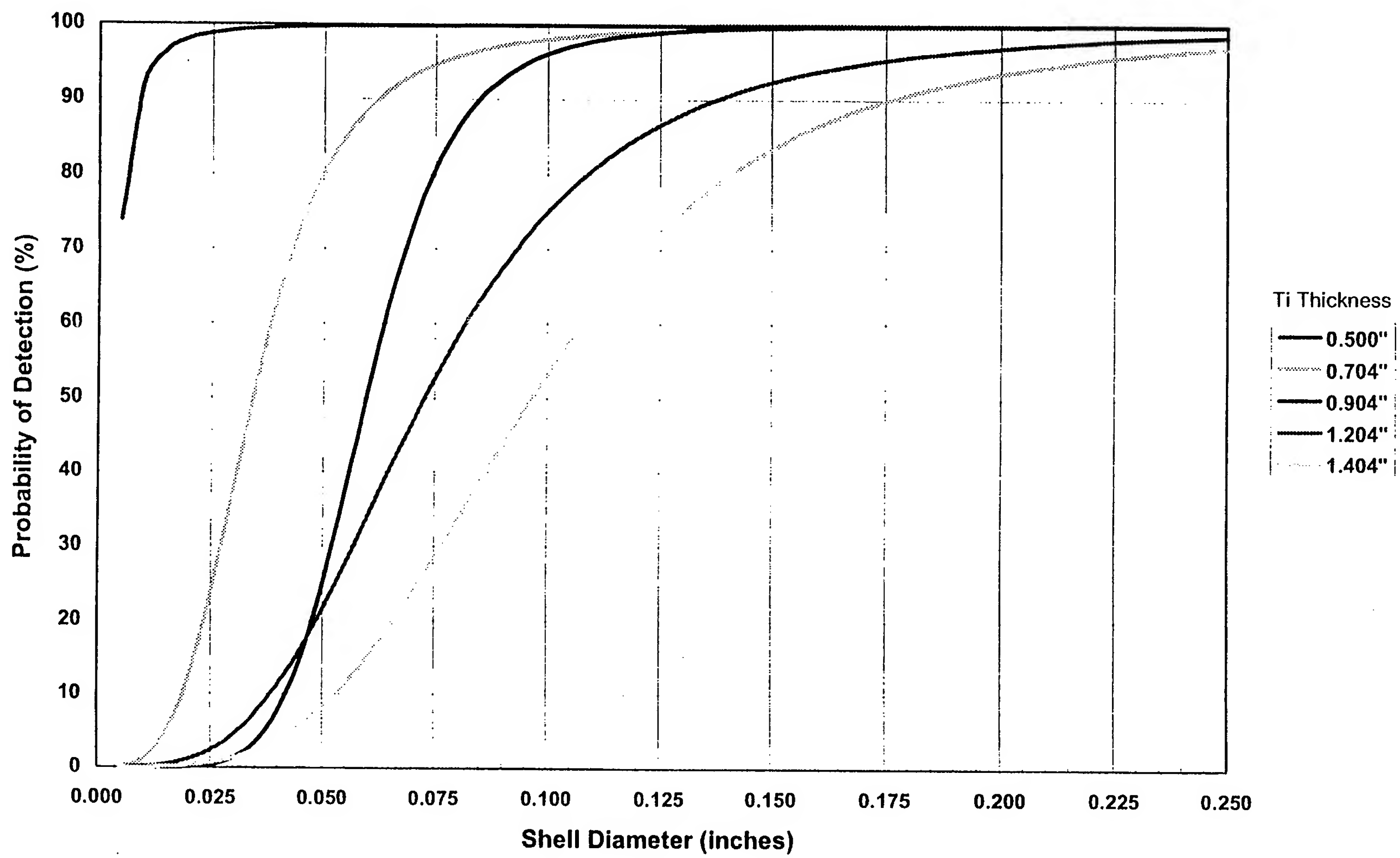
RESULTS

PoD for Shell Material A

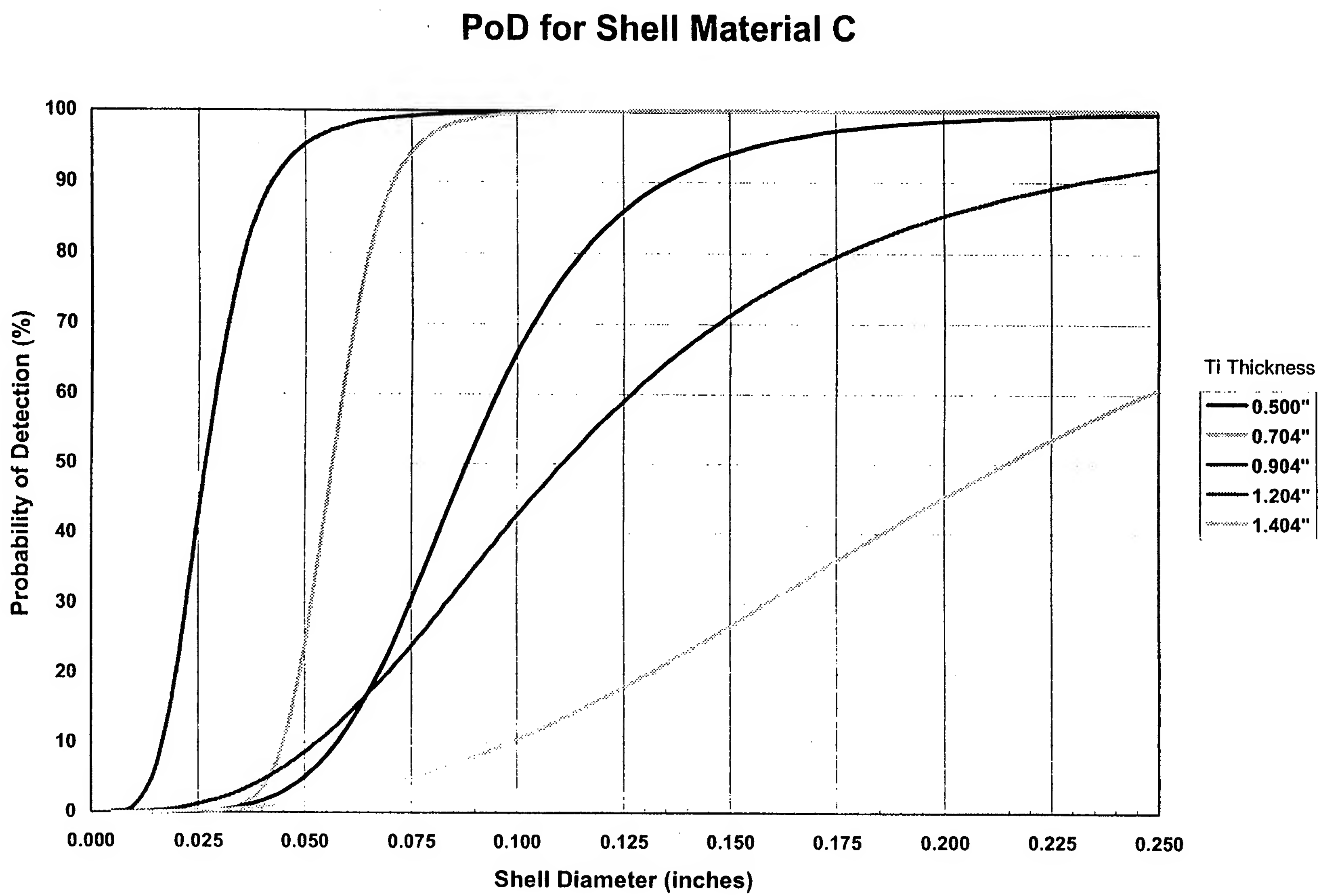


RESULTS

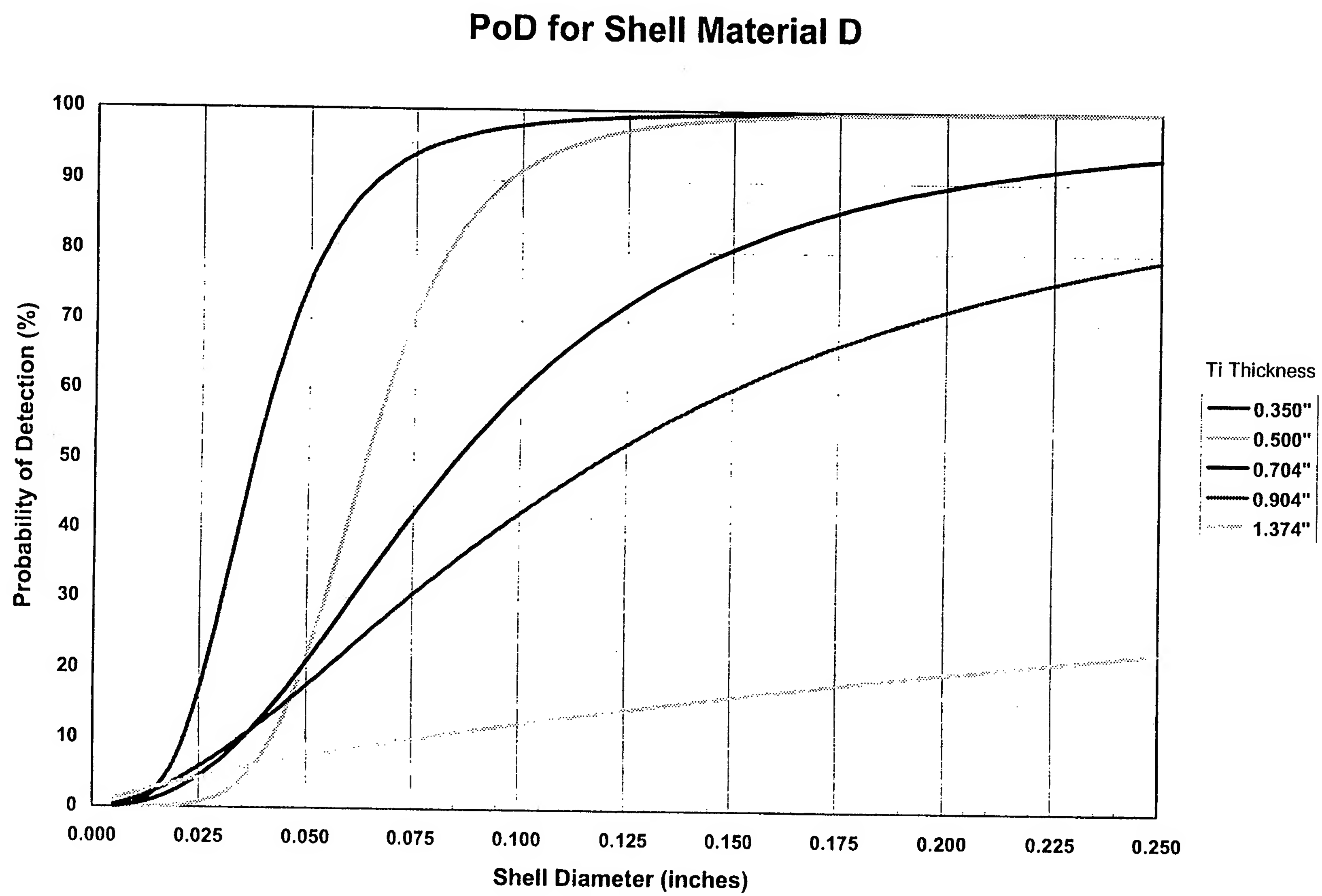
PoD for Shell Material B



RESULTS


PoD for Shell Material C
Probability of Detection (%)
100
90
80
70
60
50
40
30
20
10
0
0.000
0.025
0.050
0.075
0.100
0.125
0.150
0.175
0.200
0.225
0.250
Shell Diameter (inches)
Ti Thickness
0.500"
0.704"
0.904"
1.204"
1.404"

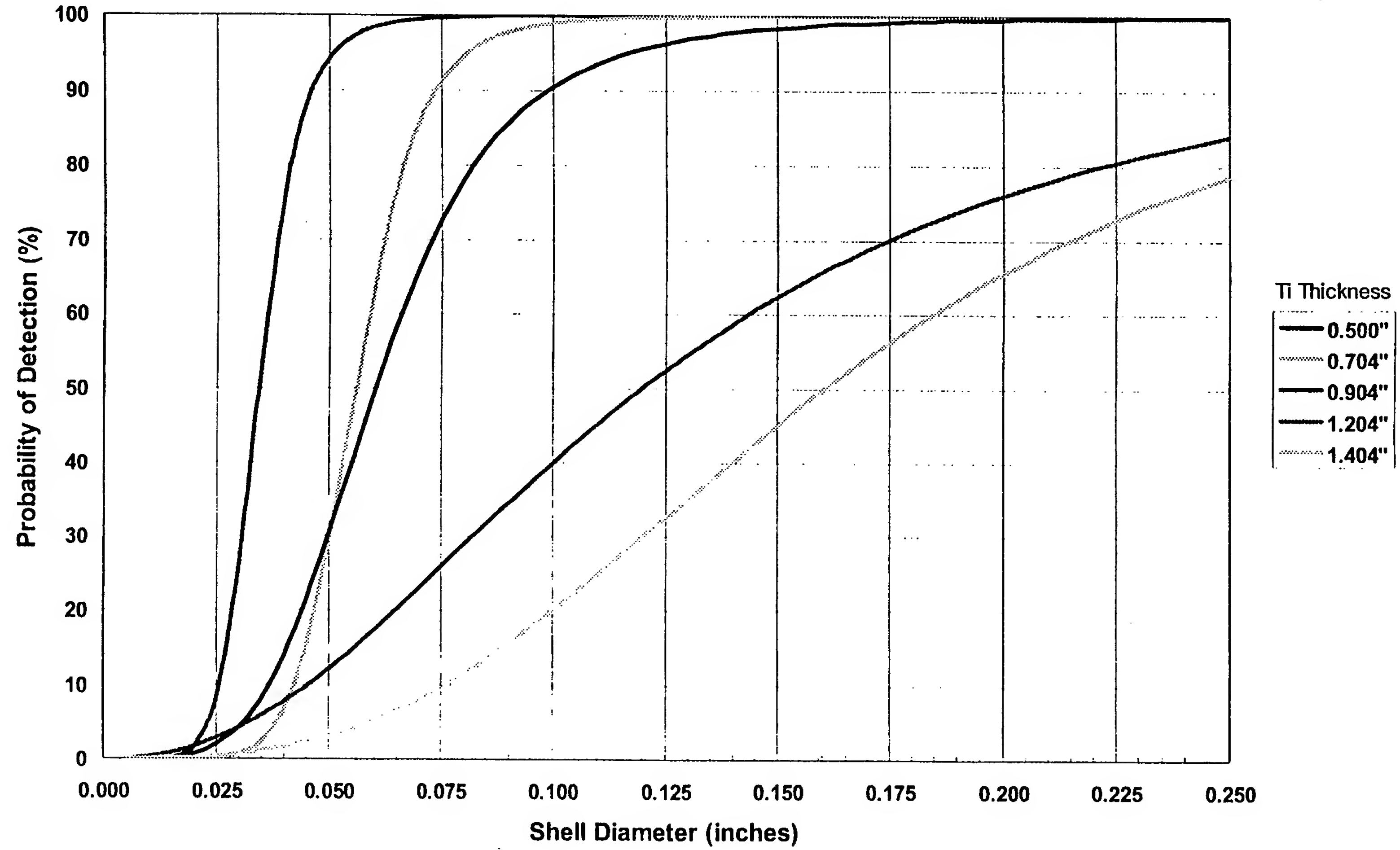
RESULTS



PoD for Shell Material D
Probability of Detection (%)
100
90
80
70
60
50
40
30
20
10
0
0.000
0.025
0.050
0.075
0.100
0.125
0.150
0.175
0.200
0.225
0.250
Shell Diameter (inches)
Ti Thickness
0.350"
0.500"
0.704"
0.904"
1.374"

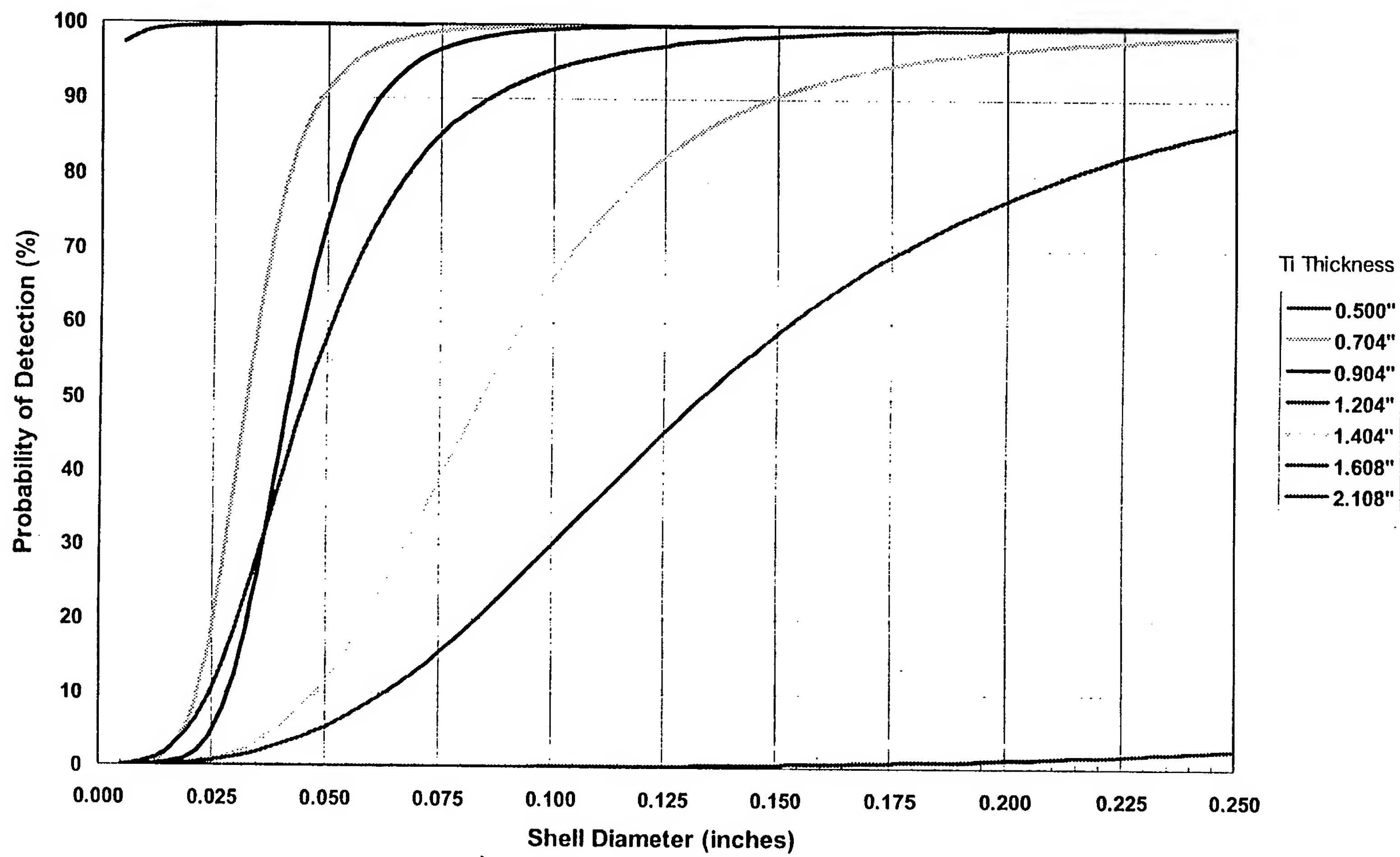
RESULTS

PoD for Shell Material E

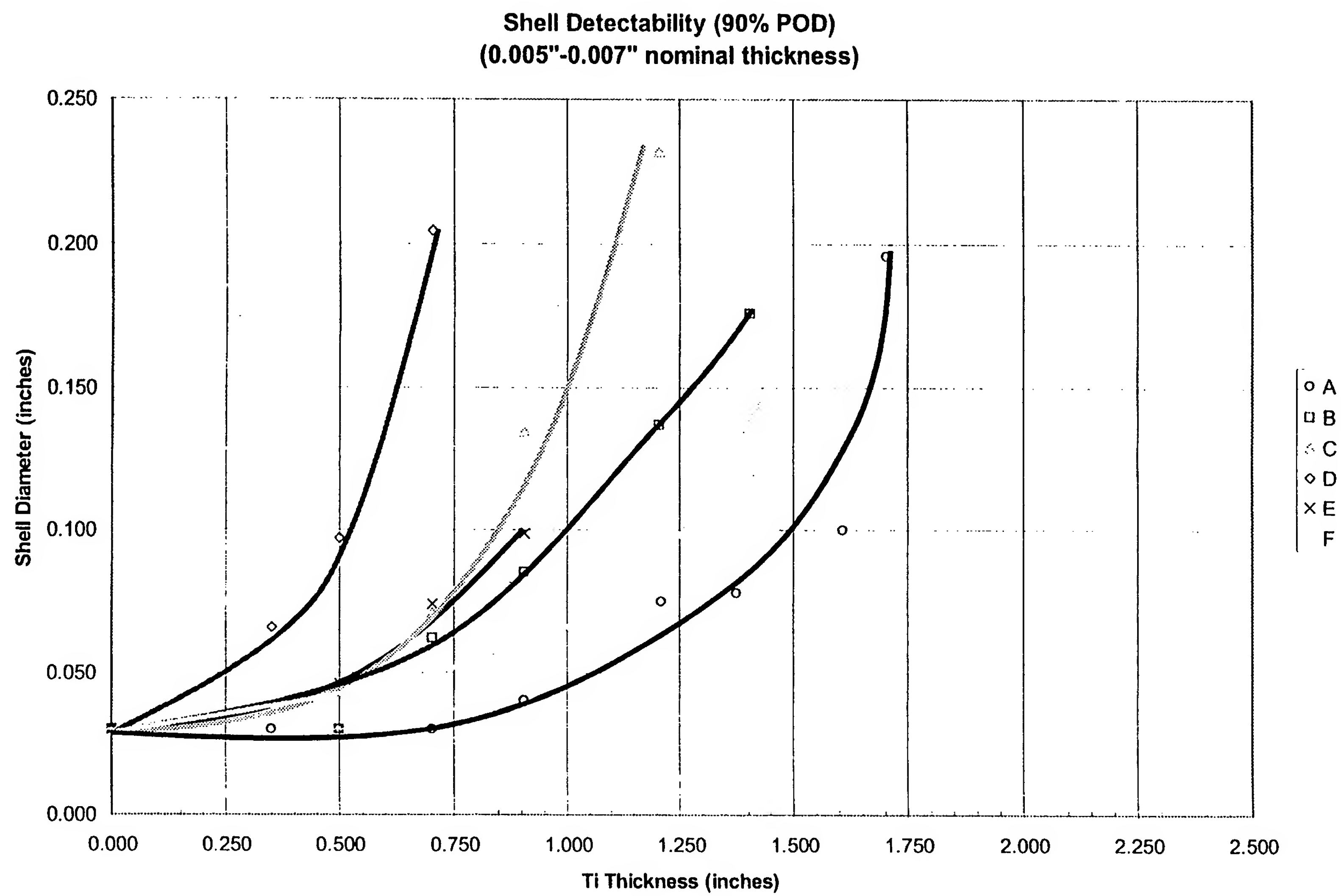


RESULTS

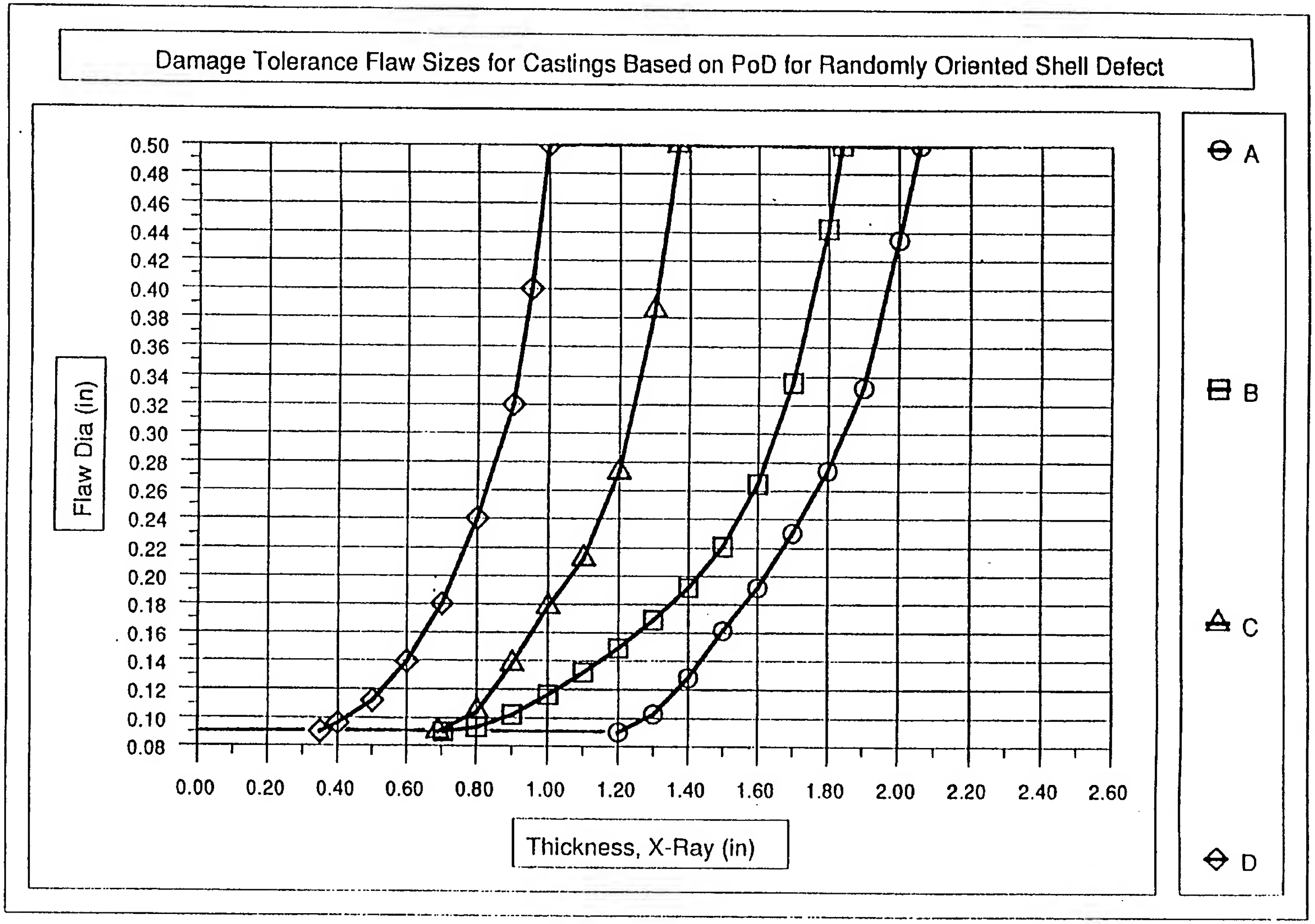
PoD for Shell Material F



90% POD VS THICKNESS



ALLOWABLES (mat'l based)



OUTPUT

- Material based acceptance criteria
- Implemented new facecoat material for enhanced detectability
- Modified inspection techniques to minimize apparent thickness
- Designs modified to accommodate detection capability limits

LESSONS LEARNED

- New applications of materials require understanding of materials processing and potential anomalies
- Effects of Defects analysis is required
- NDE capabilities and limits must be recognized & QUANTIFIED before designs are finalized
- Designs must include considerations for producibility, inspectabilitty, and maintainability

CONCLUSIONS

- Titanium HIP castings offer tremendous potential as a design option for structural applications
- Inherent producibility characteristics must be understood & incorporated into the design and inspection / acceptance program
- POD via pattern recognition has been shown effective in quantifying detection capabilities & limits for subtle density changes like inclusions
- Allowance must be made for actual anomaly size versus image size - “halo”

Advanced Ultrasonic Technique for Corrosion Detection of Aging Aircraft Structures

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ABSTRACT

Corrosion detection in aircraft structures is an ongoing challenge in terms of the availability of reliable, practical and cost effective nondestructive inspection techniques. Ultrasonic guided waves are promising but require procedure development to ensure high sensitivity and reliable transducer coupling and to provide a mechanism to transport the probe(s) over the area to be scanned.

This paper describes some practical inspection setups and procedures for corrosion detection based on guided wave transmission and reflection. It describes the results of their application to detection of disbond and corrosion in simulated and real components of aircraft fuselage. Two different lap joint constructions were assembled and subjected to accelerated corrosion in a salt fog chamber. Lap joint specimens were inspected with guided waves using a portable PC-based manual scanner. The acquired signals from a single line scans have been presented in two-dimensional format for ease of interpretation. The resulting images have been used to compare the different transducers or approaches in terms of detectability of corrosion or disbonds. The results also have been compared with those of D-Sight and eddy-current inspections performed on the same specimens.

Keywords: corrosion, ultrasonics, Lamb waves, adhesive joint inspection

INTRODUCTION

Guided waves in the form of Lamb waves are emerging as a fast and global NDT inspection technique. They demonstrate an attractive solution for disbond and corrosion detection in relatively thin plates due to their guided behavior. Moreover, with their multimode character, selection of guided wave modes can be optimized for detection of a particular type of defect. Optimization is based on analysis of the particle displacement, stress and power distributions for each mode in a structure [1,2].

Lamb wave modes have been used for the evaluation of adhesively bonded structures. Lamb wave imaging techniques [3] and tomography [4,5] were developed to facilitate interpretation and enhance inspection results. In addition, to develop an inspection procedure setup with sufficiently high sensitivity, we need to ensure a reliable constant coupling of the transducer, and provide a stable mechanism to transport the probe(s) over the scanned area.

CORROSION DETECTION WITH LAMB WAVES

Lamb modes are dispersive waves and their velocities are a function of the frequency-thickness product. Therefore, any material changes such as corrosion or lack of adhesion between two layers will affect the propagating mode amplitude, velocity, frequency spectrum and its time-of-flight.

Inspection of a lap splice joint with guided waves in a pitch-catch setup permits a guided Lamb wave mode to travel from sender to receiver producing a relatively low amplitude RF signal when a disbond exists between the two bonded parts; otherwise, it will leak into the second joint if the bond is good (Figure 1).

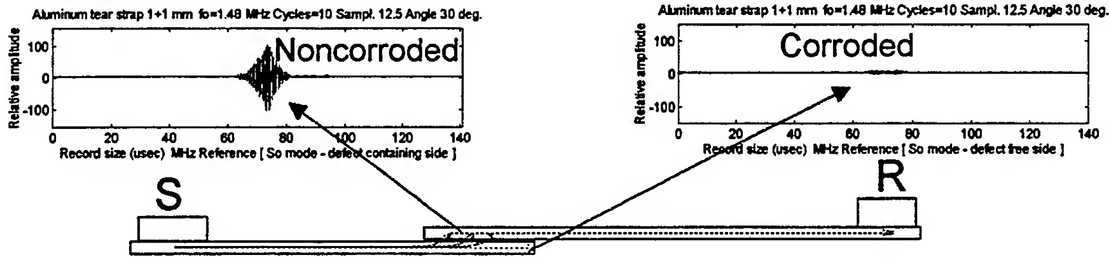


Figure 1 Transmission results from a) noncorroded area b) corroded area

In a pulse echo setup configuration, the wave will travel from sender/receiver probe producing a relatively high amplitude RF signal when a disbond exists between the two bonded layers and a low RF signal if the bond is good. Relative amplitude changes which occur in the transmitted wave through bonded/corroded structures are an indication of the existence of disbond or corrosion.

However, the overall amplitude of the RF varied greatly where ultrasonic couplant was needed in the inspection procedure. This problem was overcome by using couplant free and noncontact transducer probes.

EQUIPMENT AND INSTRUMENTATION

The system used in our experimentation was Tektrend's ARIUS Guided Wave system. This system is hosted on a manual scanner driven by an ultrasonic PC-based system. The positioning control, ultrasonic guided wave control, data acquisition, display and analysis software are all integrated into a single software package.

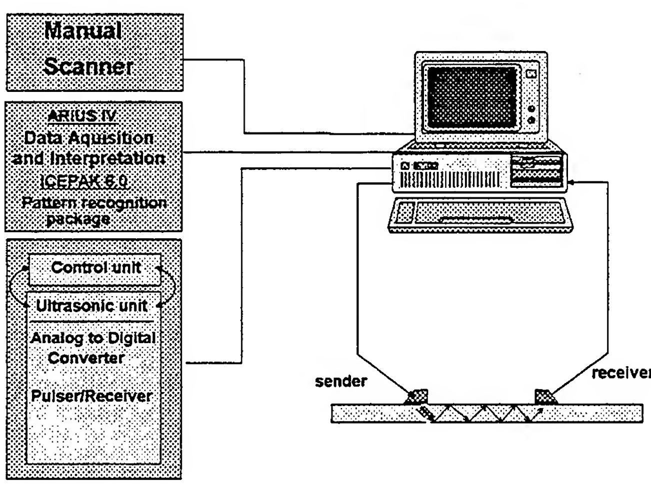


Figure 3 Integrated Guided Wave testing system

The manual Guided Wave scanning device is self contained in a single unit, in which all the electronic boards are mounted in the system computer workstation. Measurement with the above system can be made in pulse-echo as well as pitch-catch with piezoelectric transducer probes (optional with

EMAT probes). The transducer probes are driven by a toneburst pulser/receiver system. The tone burst excites a narrow-band Lamb wave mode and provides the high power to launch the wave over long distances. With tone-burst excitation, the operating frequency and the pulse characteristics of the transmitter can be controlled in a repeatable manner. The tone-burst output is fed to the transmitter as a continuous sine wave which is gated to create a burst and subsequently amplified. The signal from the receiving transducer is transferred to the broadband receiver and displayed on the ultrasonic PC-scope. Signals from each inspection scan can be stored and played back for further analysis. However, for more advanced analysis, interpretation and intelligent scans, the system contains the tools to tag signals for export to a pattern recognition package.

INSPECTION RESULTS

Inspected specimens: Tests were carried out on three lap-splice samples as shown schematically in Figure 4. These specimens were made of 1.0 - 2.0 mm thick aluminum sheets. Two of the lap joints were assembled and subjected to accelerated corrosion in a salt fog chamber. In the third lap joint, a defect was introduced between the lap layers to simulate two disbonded regions containing a 20X20 mm air gap. The dimensions of the joint plates were 400x300 mm. The width of the bonded area in a lap splice joint was typically 50:70 mm.

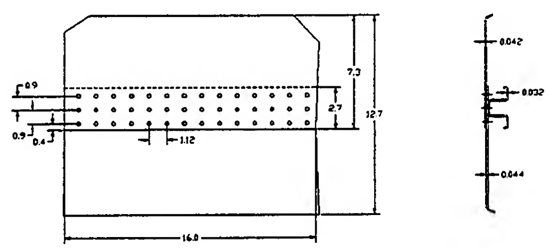


Figure 4 Inspected specimen

Comparison results: For comparison to demonstrate the efficiency of EMATs in terms of mode excitation and reception two sets of tests were performed. The figure below compares detection of a machined defect with S_0 Lamb wave in a 1.2 mm thick aluminum plate using a piezoelectric wedge transducer and an EMAT transducer.

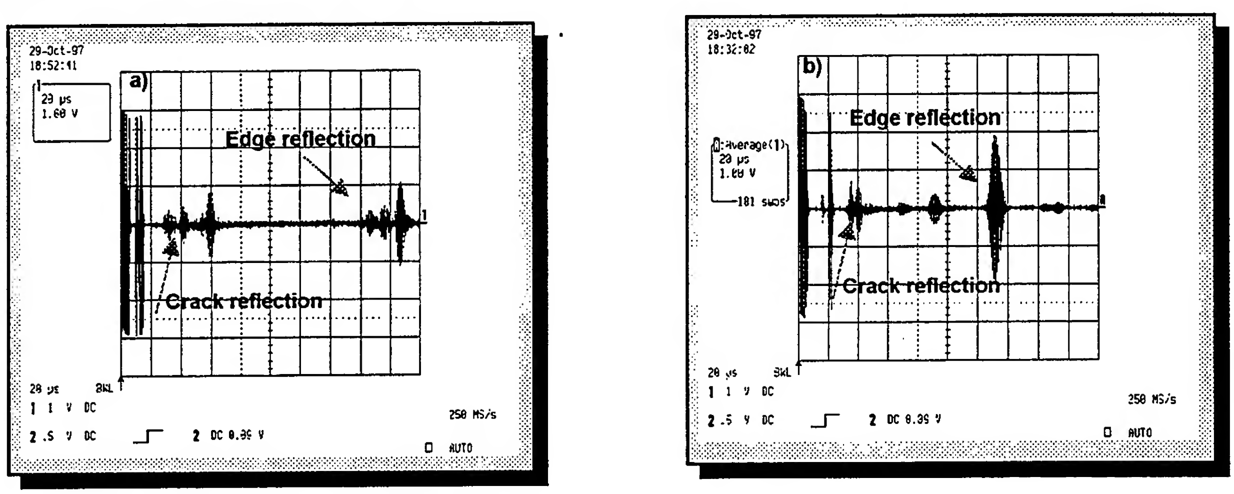


Figure 5 a) Piezoelectric RF signal b) EMAT probe RF signal

The same probes were also used to excite the S_0 mode in a 1 mm aluminum plate. The angle of the wedge probe was adjusted as well as the frequency of excitation. The EMAT probe, however, was controlled only by the frequency. Signals in the following figure show that modes excited with the EMAT probe were clearly visible and precise. However, signals from the wedge probe were associated with extra trailing edges due to reverberation in the Plexiglas wedge.

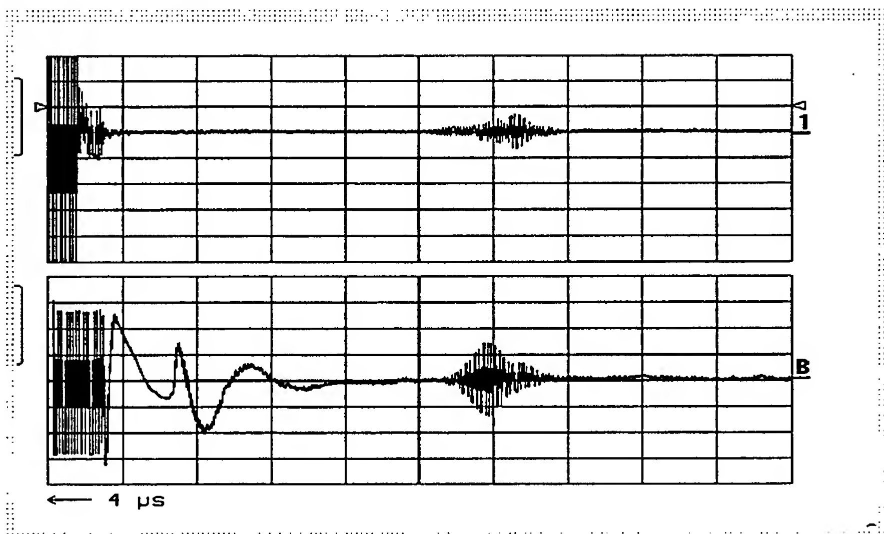


Figure 6 a) Piezoelectric RF signal b) EMAT probe RF signal

Guided Wave Inspection using Piezo-composite Probes: Guided wave inspections were performed on lap joint specimens and inspection results were evaluated in terms of the sensitivity and repeatability.

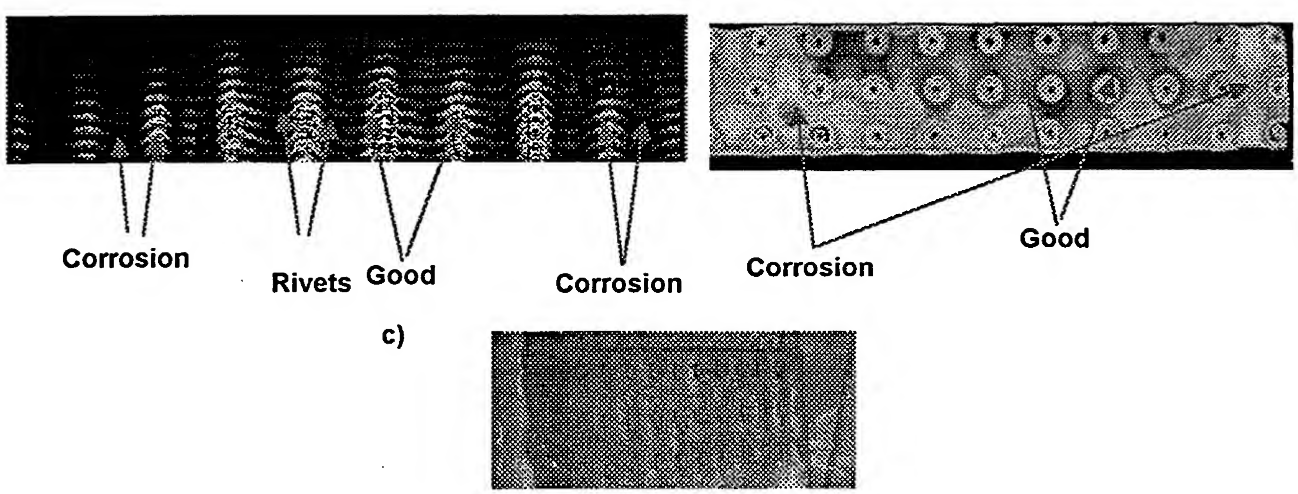
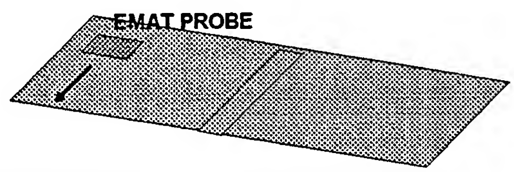


Figure 7 a) Eddy-current C-scan b) Lamb wave Scan d) D-sight image

The inspection was carried out in pitch-catch configuration using piezo-composite probes to excite the S_0 mode at 1.5 MHz. Figure 7a shows the results of this inspection and presents a series of signals in three-dimensional format. The well-bonded (non-corroded) areas are characterized by high amplitude signals (signals indicated by red colour). Poorly bonded areas (caused by corrosion) resulted in a reduction of amplitude of the received signals as shown by the low amplitude echoes at both ends of the specimen. The high and low amplitude signals are represented by the lighter and heavier colors, respectively. The interruptions between signals in Figure 5a are due to the presence of rivets.

To verify the guided wave results, these specimens were also inspected at IAR/NRC using an automated eddy-current scanner as well as an enhanced optical technique (D-Sight) [6]. Corrosion was detected in the two ends of the specimen by both techniques as shown in Figure 7b and 7c. The red and orange colors in the eddy current image show areas of severe corrosion while the green and blue represent areas having very light corrosion. In the D-Sight image, the existence of corrosion is inferred by the presence of waviness (pillowing) between the rivets which is caused by the formation of corrosion products (aluminum oxide and hydroxide) at the interface between the two plates.

Guided Wave Inspection using EMAT Probes: A second test was performed on a bonded lap joint specimen made of two 665x460x2 mm aluminum plates. The specimen contained simulated disbond areas. This specimen was scanned with the same manual scanner used in the previous inspection. However, a meander-like coil EMAT probe was used to perform the scan in pulse-echo configuration using the overlap edge of the top plate to reflect the beam (Figure 8a). The EMAT probe was designed to generate 1.5 mm wavelength Lamb waves in aluminum which corresponds to the S_0 mode at 2.52 MHz. Figure 8b shows the received signals obtained in one linear scan across the bond line. They are presented in a three-dimensional format. The blue color corresponds to low amplitude echoes or well-bonded areas where the energy of the guided waves has dissipated into the second layer. The red to light blue colors indicate high amplitude signals and disbond areas where the energy loss is substantially less. Figure 7 provides a clear indication of the presence and size of disbonds close to the edges of the lap joint.



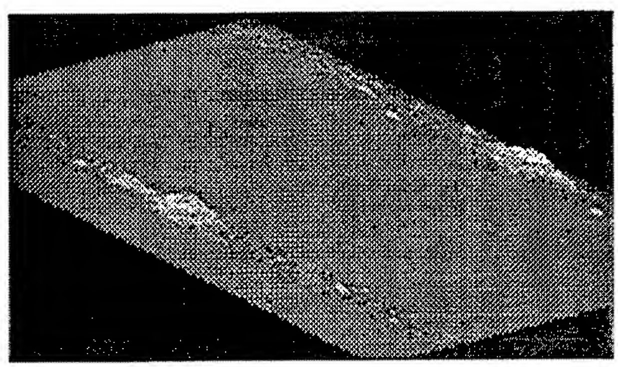


Figure 8 a) Two and three-dimensional image obtained from Lamb wave scan with an EMAT probe showing disbonds b) A schematic of pulse–echo guided wave inspection of a lap joint

The Lamb wave scan results indicate the existence of a disbond between the layers. The relatively high amplitude of the signals shown in Figure 7 represent reflections obtained from the near edge of the plate and give a clear indication of a disbond on the two sides of the joint.

CONCLUSION

A practical inspection procedure was demonstrated using Lamb waves for fast and effective inspection to detect and locate defects in layered aircraft structures. Lamb wave inspection can be carried out either by using two probes in pitch-catch or one probe in pulse-echo configurations. It can detect disbonds in lap-splice joints in a single scan and the procedure setup is suitable for presentation of the results as an image relating the amplitude and time-of-flight to facilitate interpretation. Both piezo-composite and EMAT probes can be used, however, EMAT probes are more efficient in terms of mode excitation and reception. Furthermore, different Lamb wave modes can easily be generated by controlling the excitation frequency of the EMAT probes. These probes also show less sensitivity to surface paint or roughness as compared to the conventional piezoelectric transducers. In terms of scanning, no physical contact with the test structure or fluid coupling is necessary for the EMAT probes. This makes the EMAT-based guided wave technique attractive for high speed large area inspections in the field.

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[5] D.P. Jansen and D.A. Hutchins, "Lamb Wave Tomography," IEEE Ultrasonics symp.,1990 p1017-11020.

[6] A. Fahr, D.S. Forsyth, C.E. Chapman and R.W. Gould,"Advanced ultrasonic methods for Airframe Corrosion Detection," SMPL/ IAR, NRC Final report submitted to Tektrend Intl. Inc under a contract from Defence Industry Research Program.

DEVELOPMENT OF AN ELECTROCHEMICAL SENSOR FOR EARLY DETECTION OF FATIGUE DAMAGE IN AIRCRAFT STRUCTURES

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Southwest Research Institute

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Tensiodyne Scientific Corporation

AFRL Program Manager: Tobey Cordell
SA-ALC POC: Jimmy Turner

1997 USAF Aircraft Structural Integrity Conference
December 2-4, 1997
San Antonio, TX

Overall Program Objective

Assess feasibility of EFS technology for improving durability and damage tolerance assessments of military aircraft through early detection of fatigue damage -- both prior to and after the onset of cracking.

Stages of EFS Application

- **Stage 1**: Laboratory tool to characterize and understand fatigue damage evolution in controlled experiments
- **Stage 2:** Depot tool for periodic inspection for extent of fatigue damage in aircraft
- **Stage 3:** In-service tool for continuous monitoring of fatigue damage evolution in aircraft during flight

Focus of Current Paper

- Electrolyte optimization for aircraft alloys
- Influence of electrolyte on S-N behavior
- EFS response in aircraft alloys
- Technical challenges overcome
- Assessment of remaining challenges

T-38 Aircraft Alloys Under Study

Alloy	Yield Strength
7075-T7351	64 ksi
AISI 4130	149 ksi
Ti-6Al-4V	130 ksi

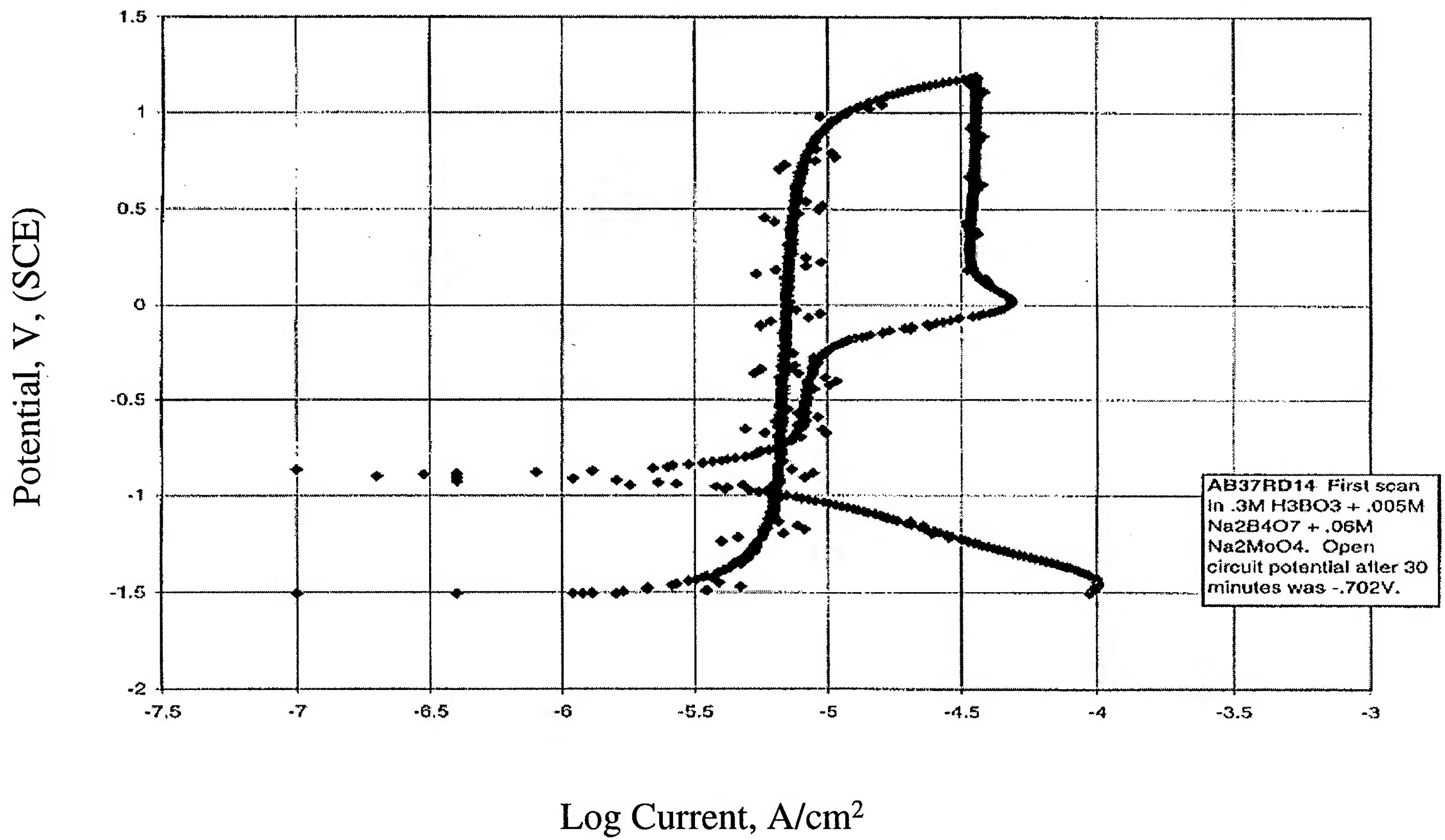
Electrolyte Optimization

Objective: Develop an electrolyte for each alloy which will enable EFS measurements without deleterious effects to the alloys -- or aircraft.

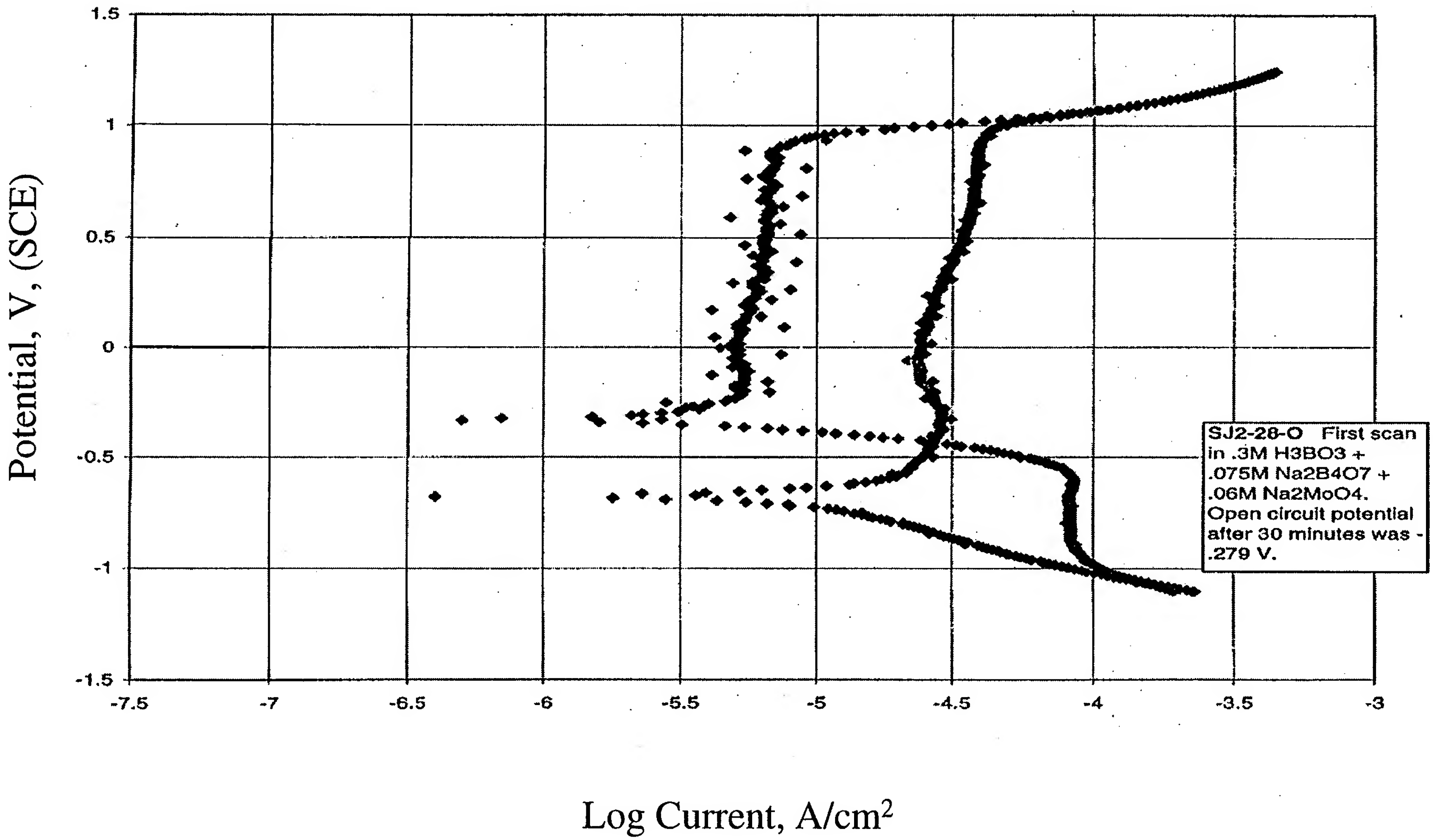
Approach: Explore candidate electrolytes using:

* Electrochemical polarization tests
* Immersion tests
* Microscopic exam. to preclude pitting
* S-N tests in the selected electrolytes

Cyclic Polarization Curve: 7075



Cyclic Polarization Curve: 4130



Immersion Test Results

Electrolyte:

0. 3M H_3BO_3 + 0.03M $Na_2B_4O_7$ + 0.06M Na_2MoO_4

Alloy	Weight Loss (mm/yr)*
7075	0.014
4130	0.0019

* *Ave. from 4 samples each in 10 and 20 day tests*

General Corrosion Index

Corrosion Index	Weight Loss (mm/yr)
Outstanding	< 0.02
Excellent	0.02 to 0.1
Good	0.1 to 0.5
Fair	0.5 to 1
Poor	1 to 5
Unacceptable	> 5

Optimized Electrolytes

Aluminum Alloys:

0. 3M H_3BO_3 + 0.005M $Na_2B_4O_7$ + 0.06M Na_2MoO_4

pH = 6.9

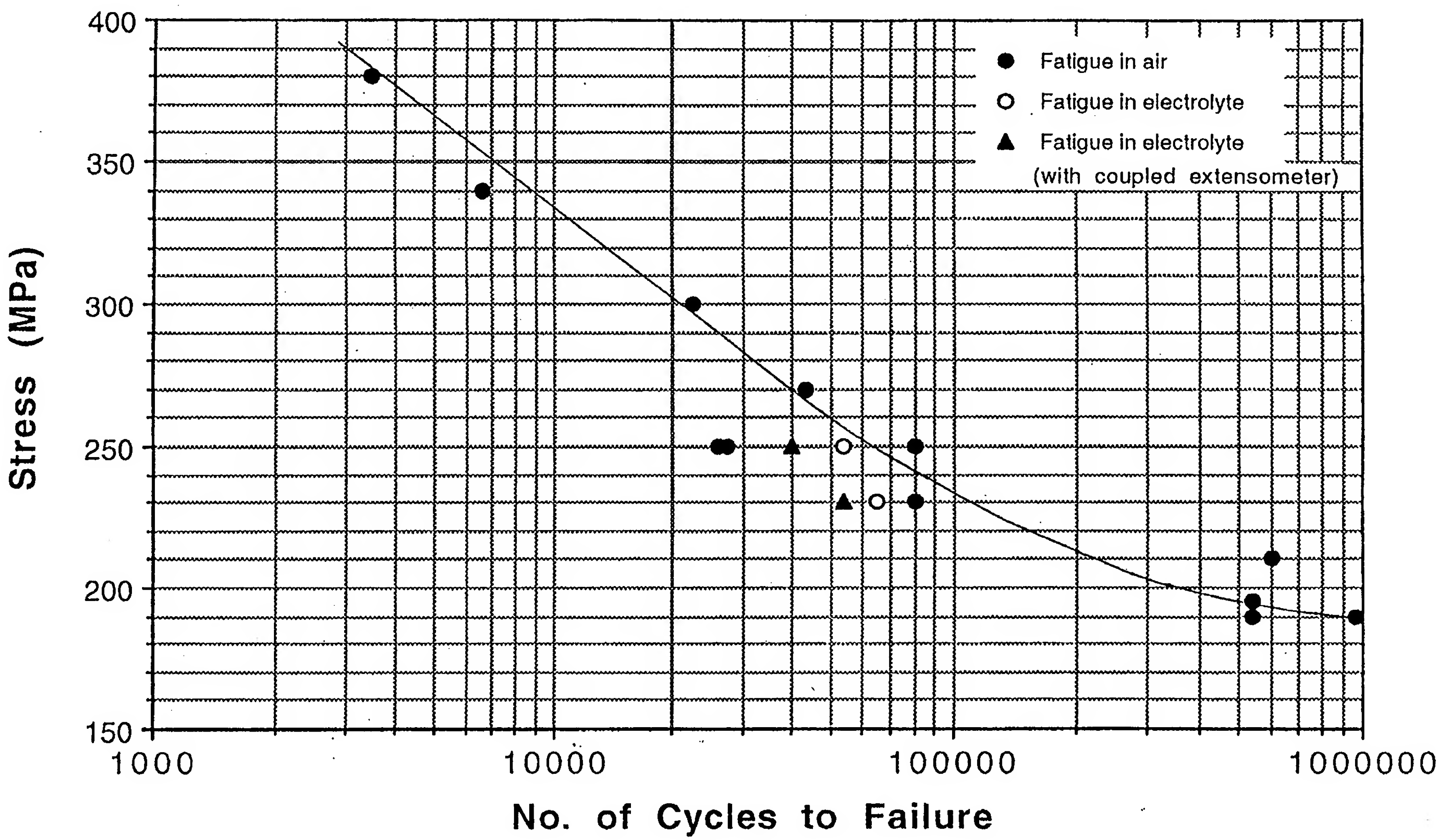
Steels:

0. 3M H_3BO_3 + 0.075M $Na_2B_4O_7$ + 0.06M Na_2MoO_4

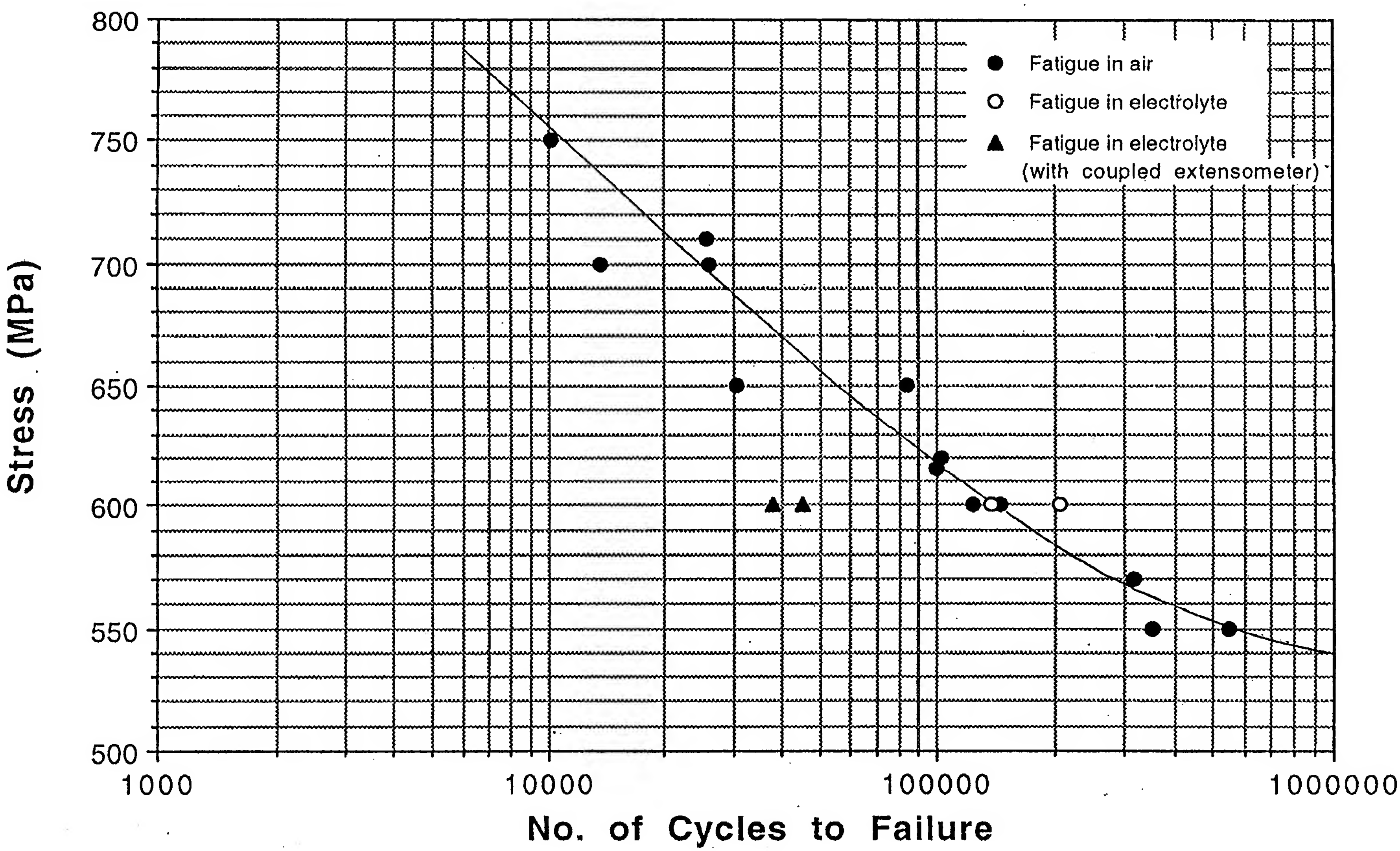
pH=8.4

Note: *pH adjusted to insure passivation and preclude pitting*

7075 S-N Behavior: Air vs. Electrolyte

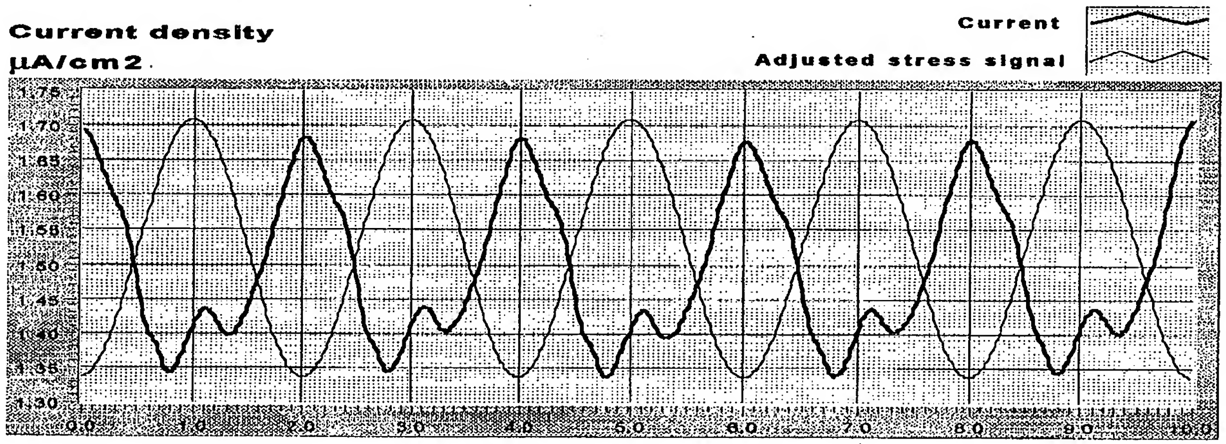


4130 S-N Behavior: Air vs. Electrolyte

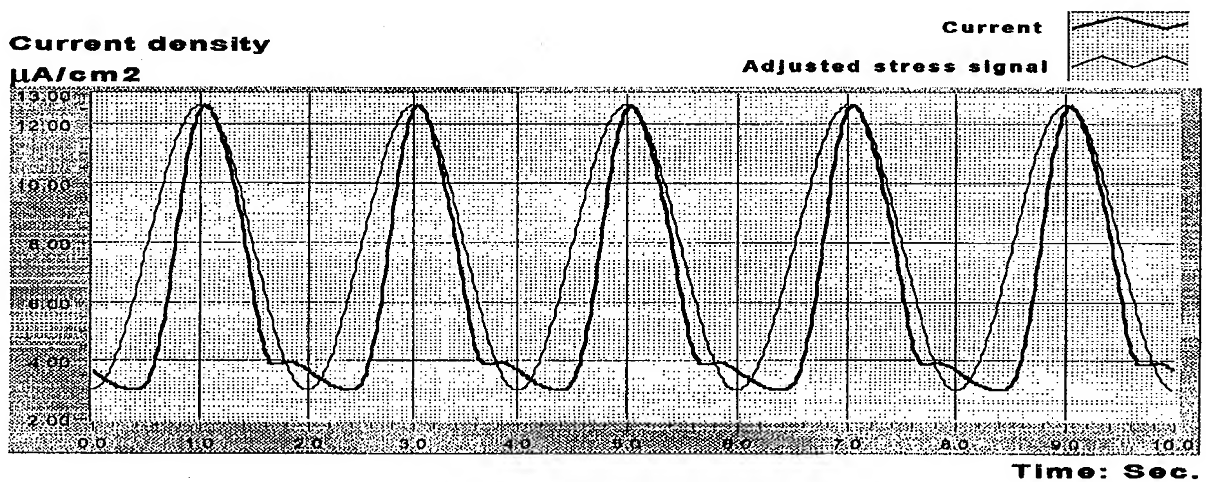


EFS Phase Relationships

Early-Life
1.9 kC



Late-Life
190 kC

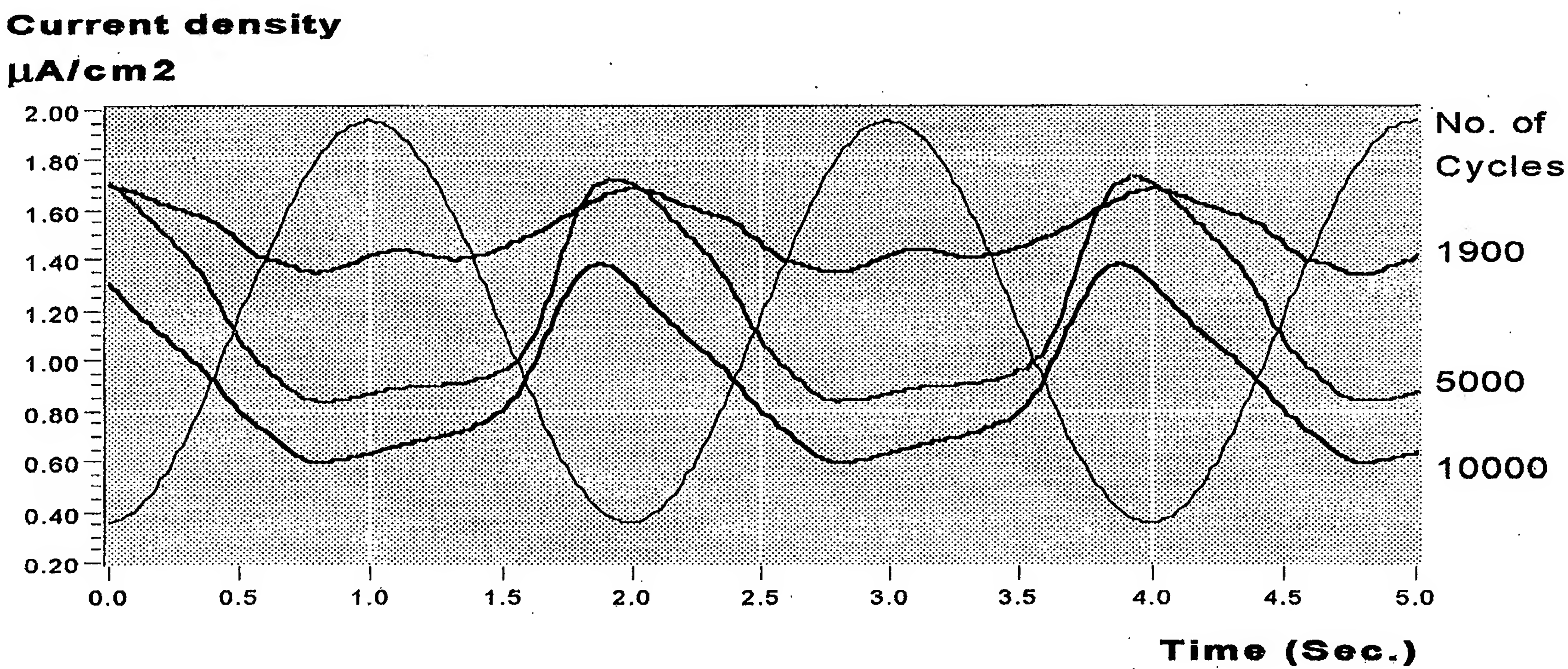


Transient current of a 4130 Steel speciman Cycled at 600 MPa (0.5 Hz) in 0.3M H_3BO_3 + 0.075M $Na_2B_4O_7$ + 0.06M Na_2MoO_4 solution, (pH=8.4). Applied potential = 0.4 V (SCE). Recorded at 204,000 cycle.

EFS Current vs. Load Transients

Early Life (initial 5%) in 4130 Steel

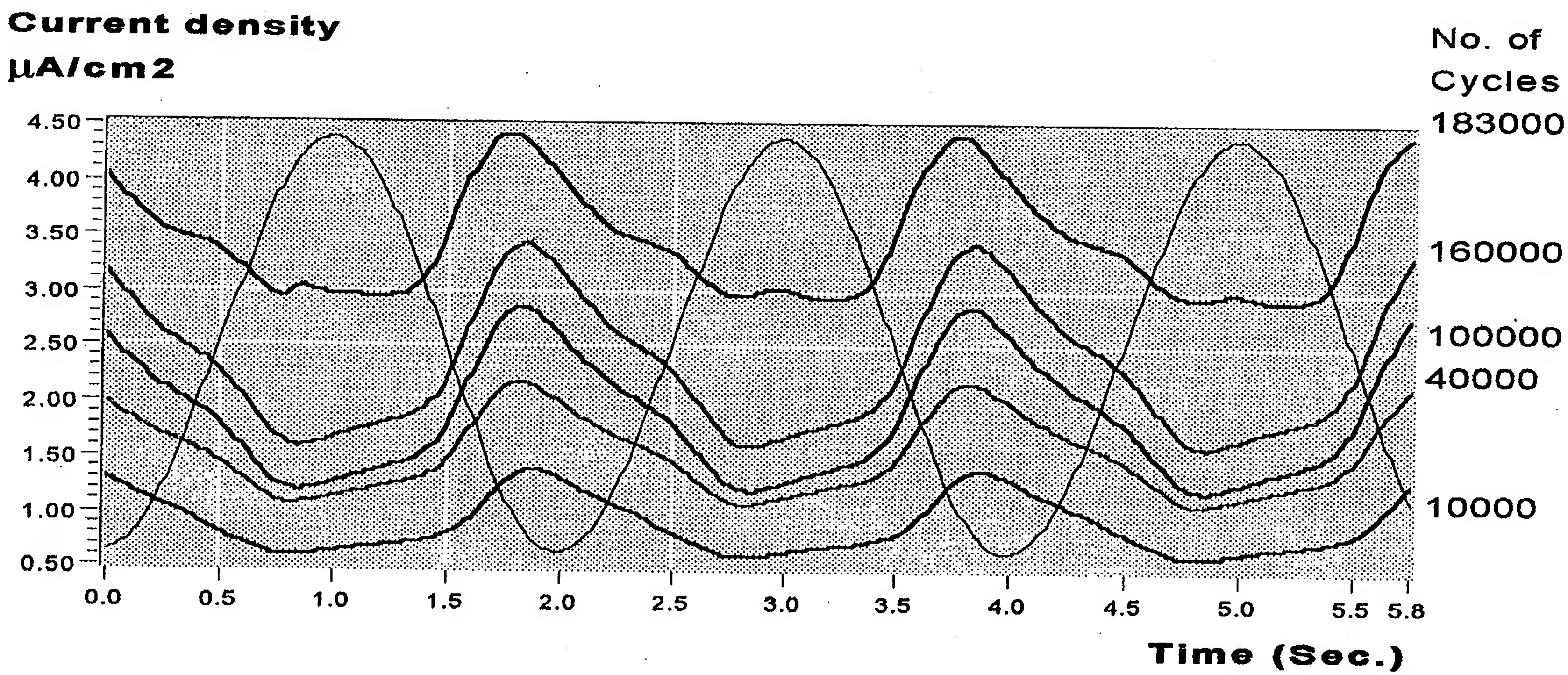
$\Delta\sigma = 600$ MPa R=-1



EFS Current vs. Load Transients

Mid-Life (to 80%) in 4130 Steel

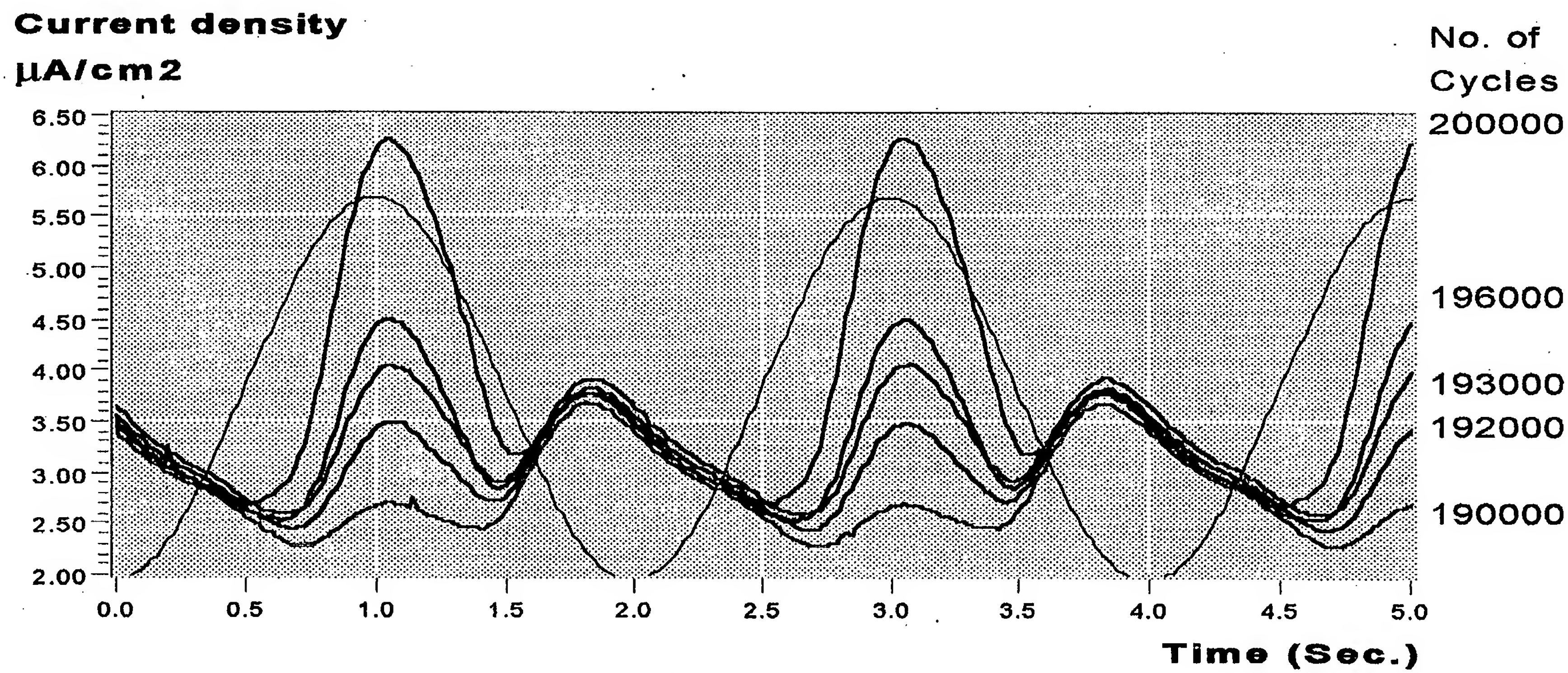
$\Delta\sigma = 600$ MPa R=-1



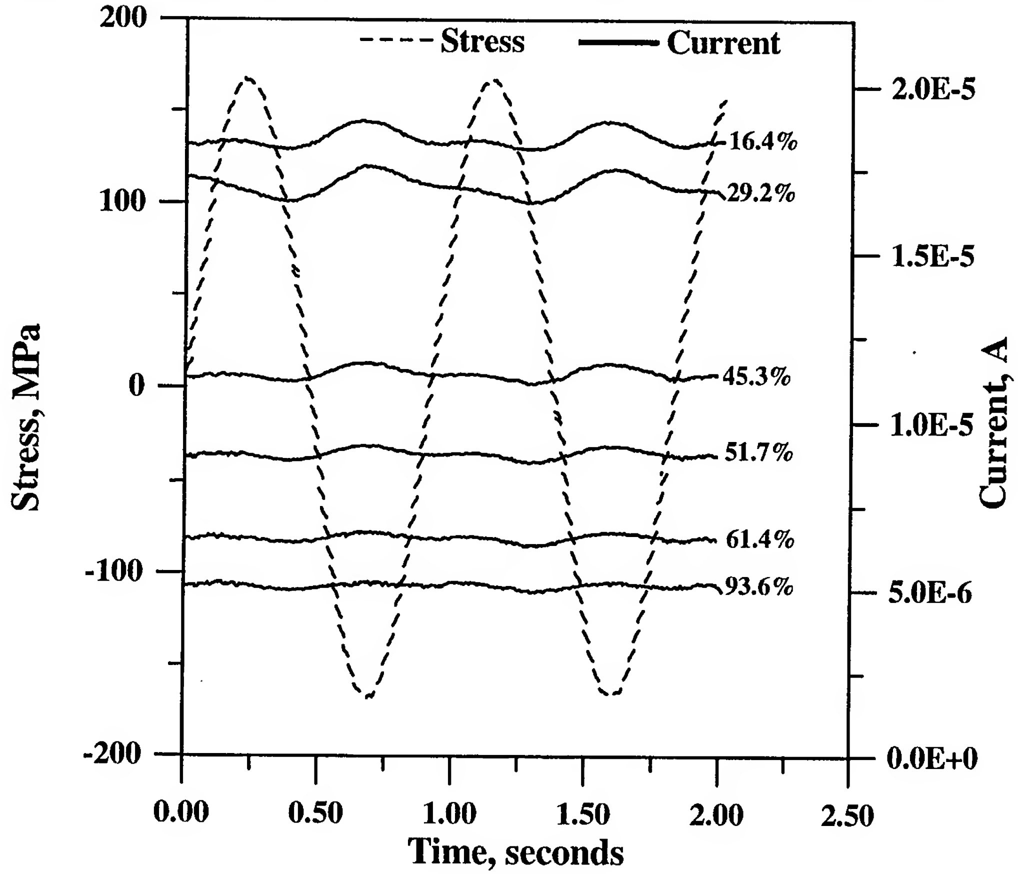
EFS Current vs. Load Transients

Late-Life (last 20%) in 4130 Steel

$\Delta\sigma = 600$ MPa R=-1

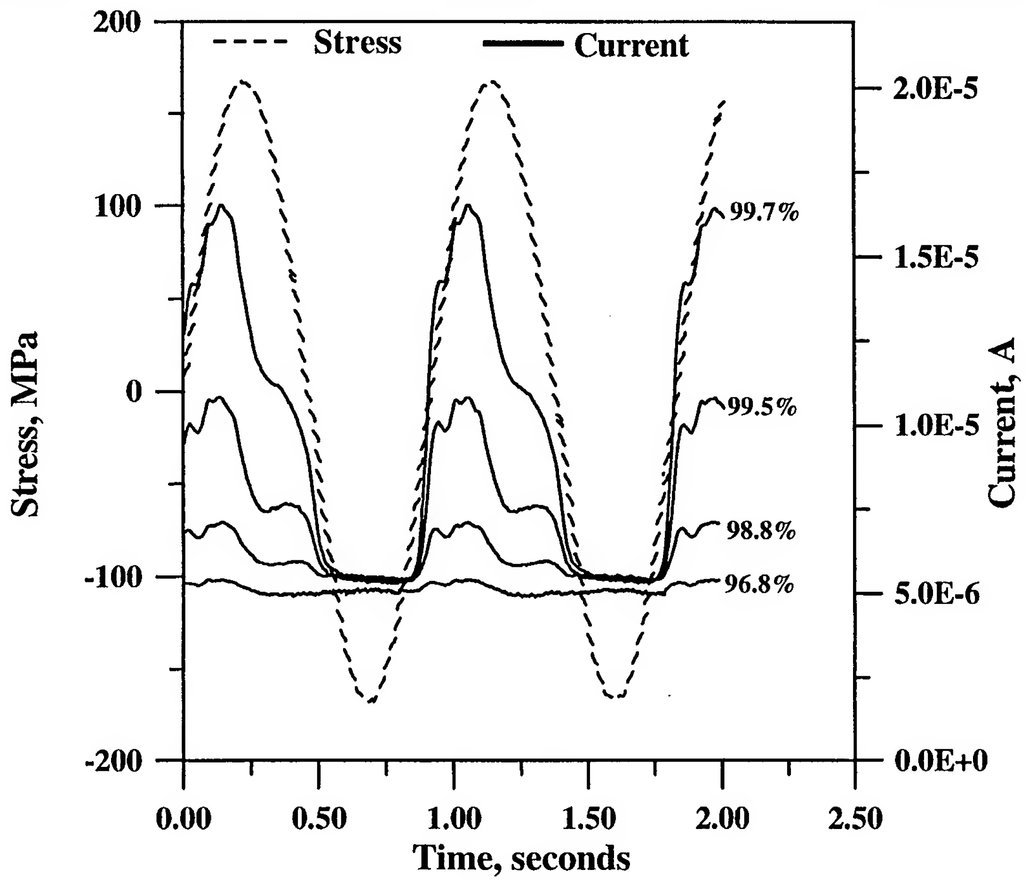


EFS Current vs. Load: 7075 Al Early Life



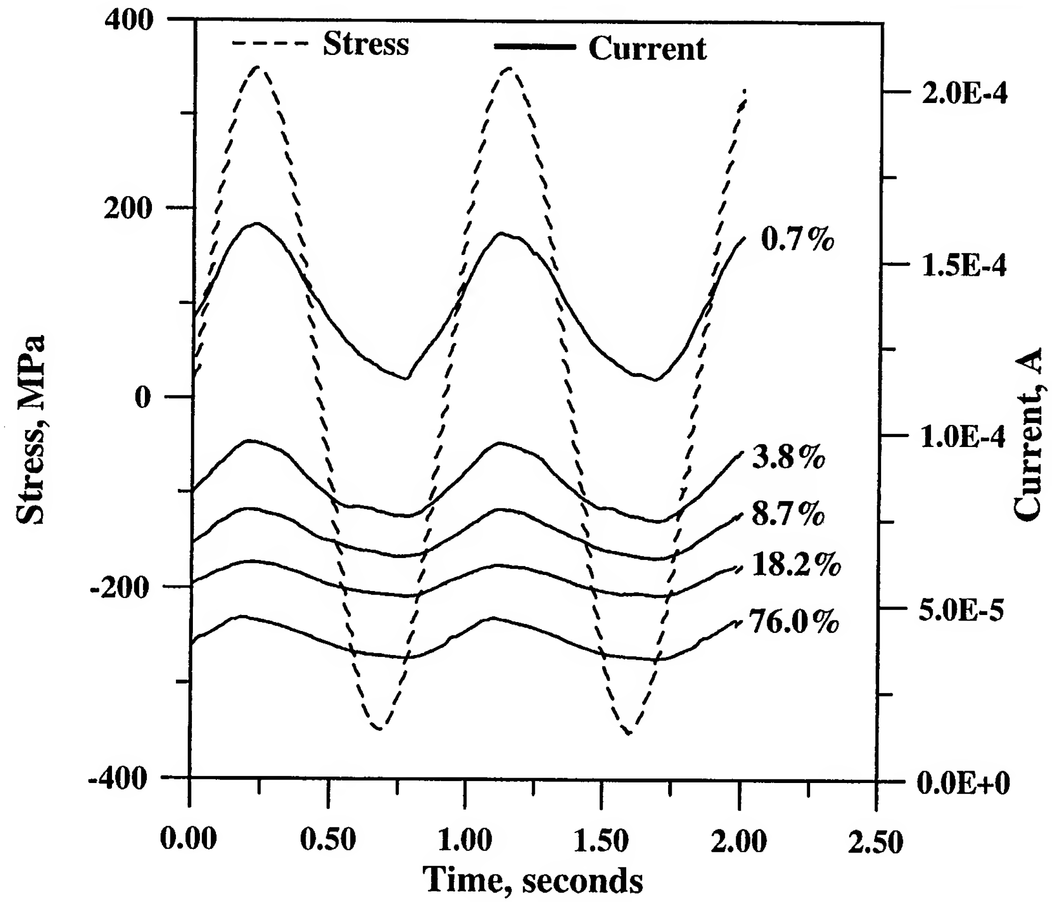
- 7075 Al
- 23 ksi (159 MPa)
- R = -1
- 300 mV vs SCE
- Failure at 310,840 cycles

EFS Current vs. Load: 7075 Al Late Life



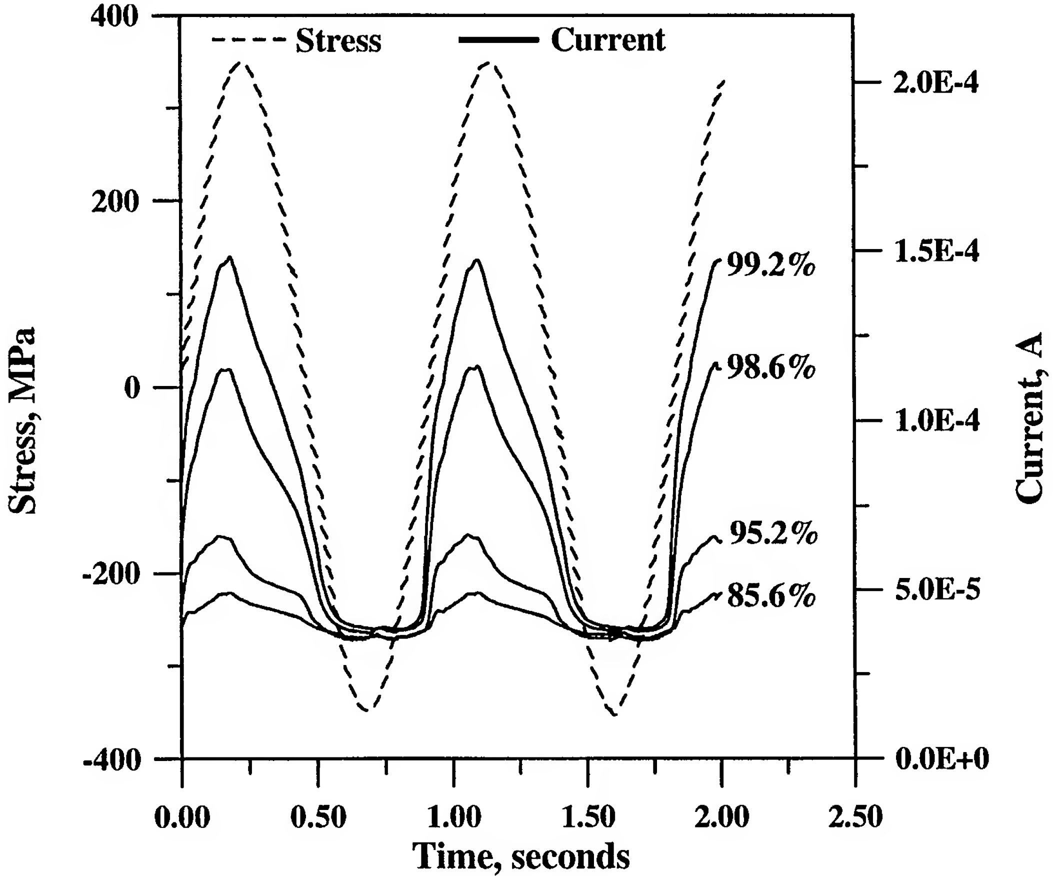
- 7075 Al
- 23 ksi (159 MPa)
- R = -1
- 300 mV vs SCE
- Failure at 310,840 cycles

EFS Current vs. Load: 7075 Al Early Life



- 7075 Al
- R = -1
- 50 ksi (345 MPa)
- 300 mV vs SCE
- Failure at 10,397

EFS Current vs. Load: 7075 Al Late Life



- 7075 Al
- R = -1
- 50 ksi (345 MPa)
- 300 mV vs SCE
- Failure at 10,397 cycles

Summary & Conclusions

- Electrolyte successfully developed having
  - negligible general corrosion
  - little influence on S-N behavior
  - need to examine potential for crevice corrosion
- EFS response demonstrated for typical aircraft alloys in optimized electrolyte
- EFS response reproduced at Penn & SwRI
- EFS response changed systematically with increasing fatigue damage

Remaining Technical Challenges

- Electrode optimization
- Viability of EFS for coated structures
- Calibrating EFS response to fatigue damage
  - loading variables: N_f ($\Delta\sigma$, R), cyclic freq.
  - spectrum loading: simple, complex
- Improving signal/noise for field meas.
  - Passive: sensor design
  - Active: signal processing

Electrochemical *In-Situ* Sensors for Detecting Corrosion on Aging Aircraft

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Phase II SBIR funded by AFOSR (Maj. Hugh DeLong)

Objective

DSI

- To develop nondestructive means of detecting early stages of coating degradation and substrate corrosion
  - Field compatible
  - Low cost

Motivation

DSI

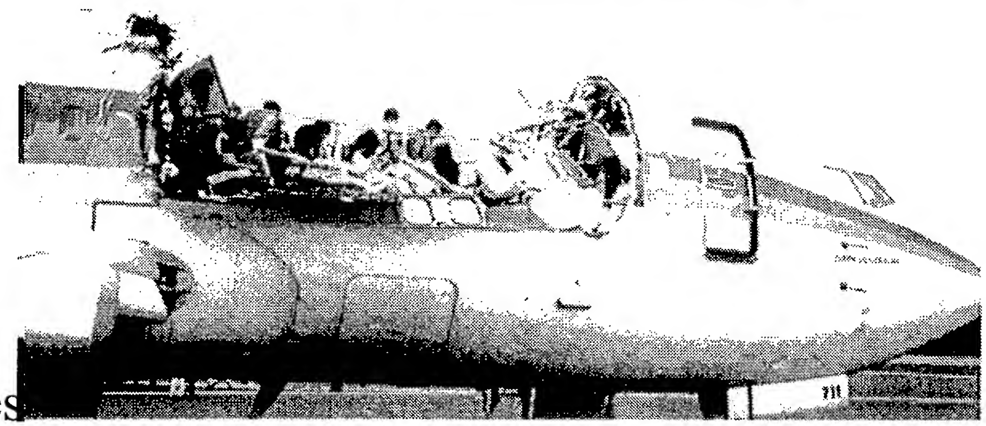
- Military and commercial air fleets and infrastructure are aging and only slowly being replaced

	Total	*Average Age (1993)*	*Projected Retirement*
A-10	222	11.2	
B-52	148	31.4	2030
C-5A	77	25	2021
C-9	35	21.5	
KC-10	59	7.7	
C-130	334	21.9	2030
C-135	479	30.9	2040
C-141	241	26.9	2010
F-15	688	8.3	2020
F-16	866	3.7	2020
F-111	232	21.5	
T-37	504	29.8	
T-38	685	26.1	
Total	*4493*	*18.6*	

Corrosion and Disbonds Can Be Very Serious

DSI

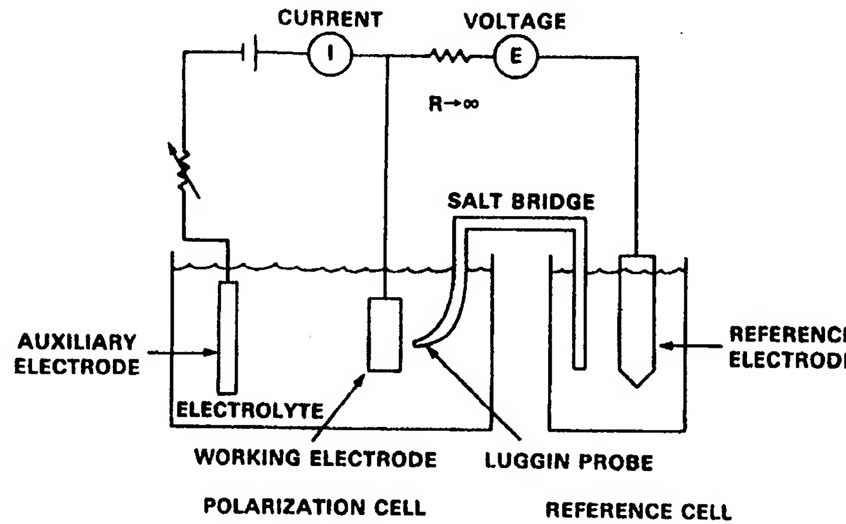
- Aloha 737 accident
  - Multisite damage (MSD) resulted from disbonding and corrosion
- Hoeppner
  - 11 military and 70 civilian fatalities in 17 yr period
- ASA Embracer 120
  - Undetected corrosion of propeller caused 10 fatalities
- Stavraopol An-24
  - Corrosion allowed disintegration in air with 50 fatalities



Background -- Electrochemical Impedance Spectroscopy

DSI

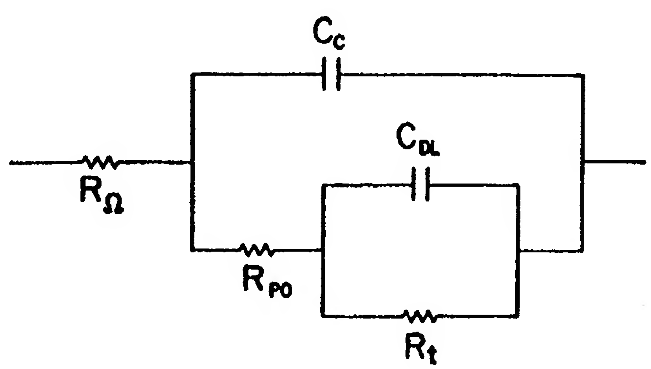
- EIS involves applying a small ac voltage to specimen and monitoring its response over a wide range of frequencies
- EIS is used to detect coating deterioration and substrate corrosion

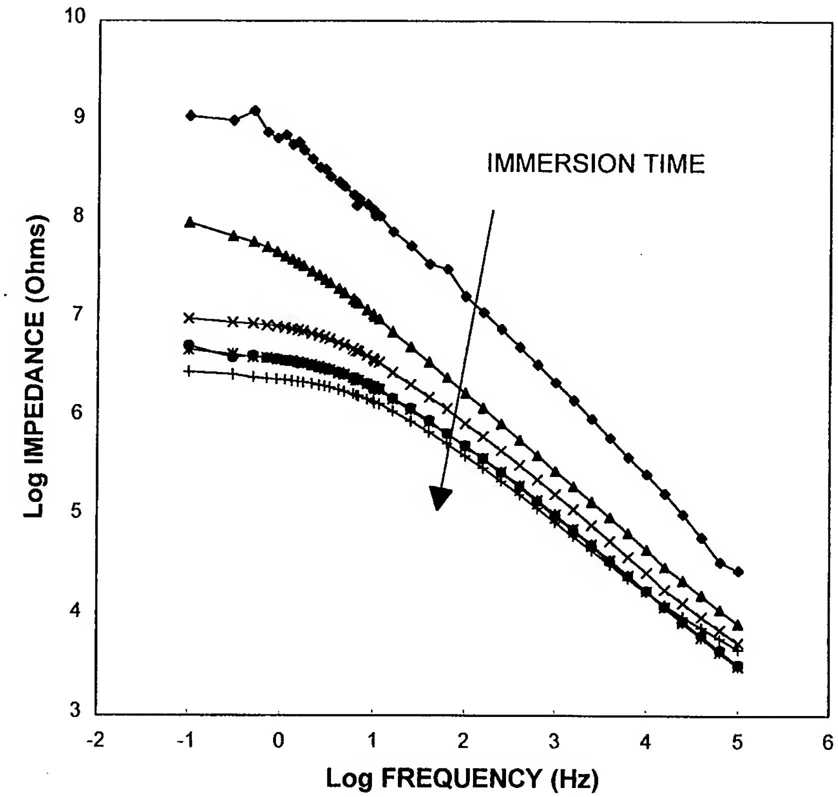


Background -- Electrochemical Impedance Spectroscopy

DSI

- During coating degradation or corrosion, current between working and counter electrodes increases (impedance decreases, especially at low frequencies)
- Coated specimen can be modeled as an electric circuit



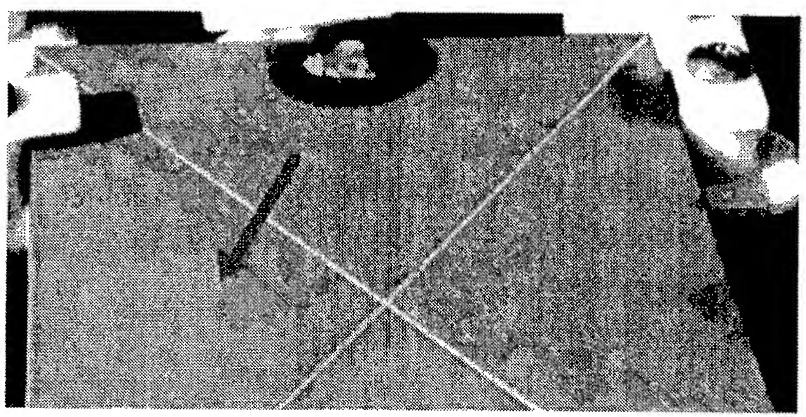


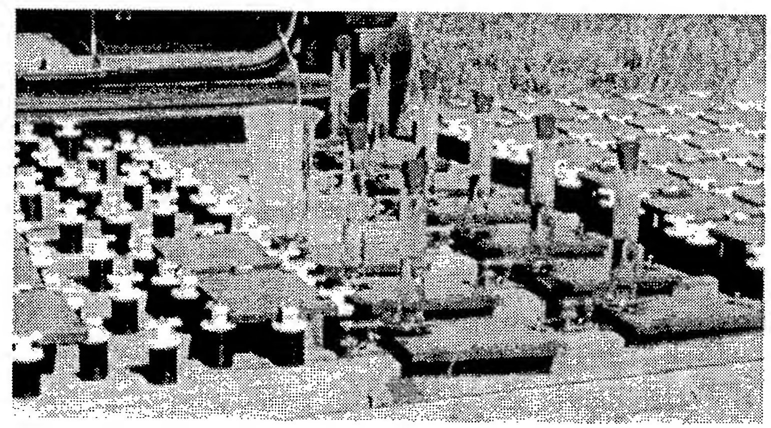
Sensor is Major Improvement Over Conventional EIS

DSI

- DSI Corrosion Sensor
  - Field or test chamber or immersion
  - Permanent electrode for inaccessible regions or portable electrode for convenient areas
  - Arbitrary structure configuration
  - Detection area can be large

- Conventional EIS
  - Requires immersion or clamp-on liquid cell
  - Cell requires accessible, flat, horizontal area and messy electrolyte
  - Coating is monitored only where exposed to electrolyte
  - Coating can suffer artifactual damage

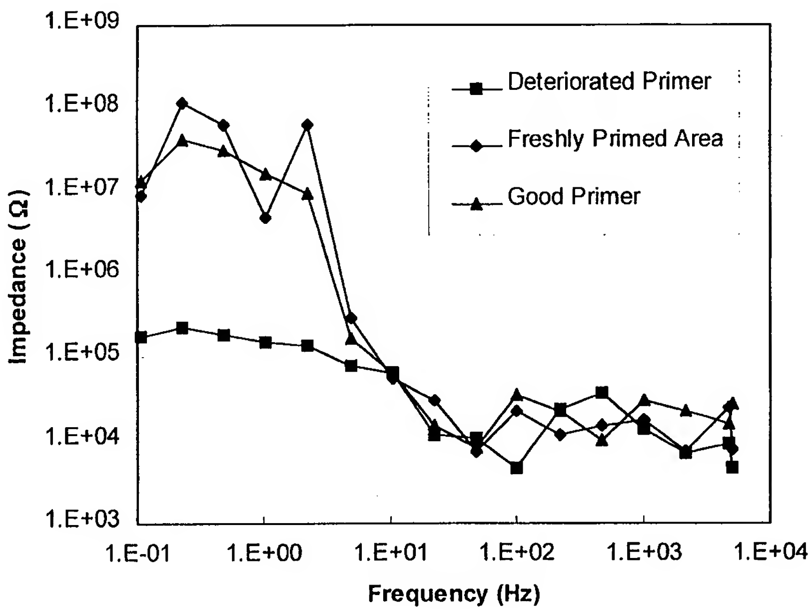


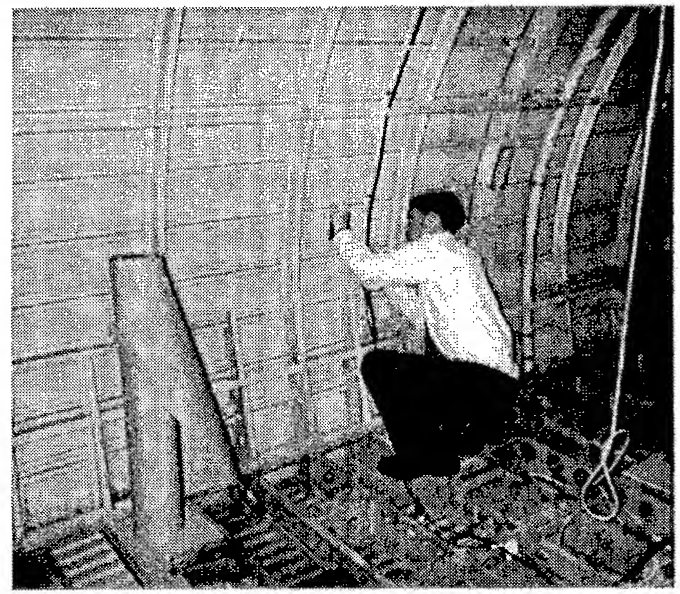


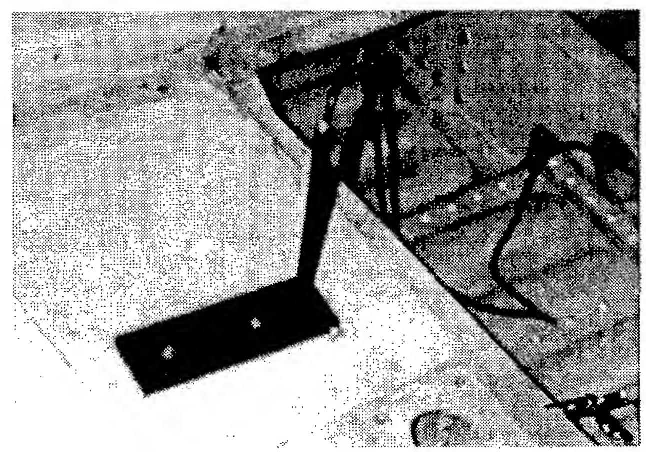
Inspection of Aircraft During Maintenance

DSI

- Inspection of C-135 aircraft clearly distinguished between good and deteriorated primed areas







Sensor Comparison

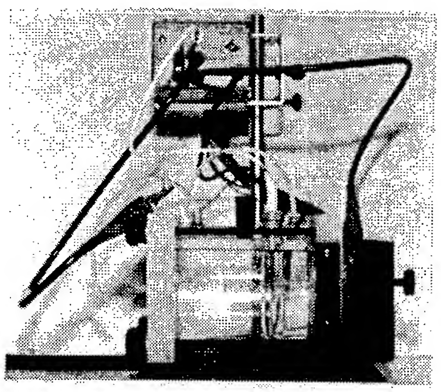
DSI

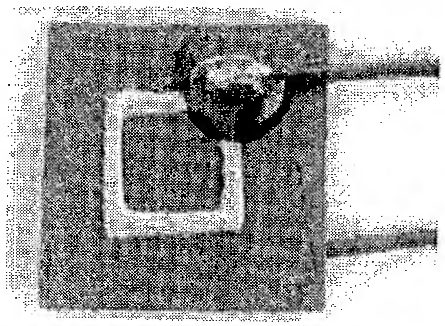
- DSI Corrosion Sensor
  - Measures corrosion of actual structure
  - Sensitive to early stages of corrosion/degradation
  - Very sensitive to moisture intrusion into bondline
  - Relatively inexpensive instrumentation
  - Monitors electrochemical process (corrosion) directly
- Other Corrosion Sensors
  - Time of wetness monitors
  - Corrosion of sensor itself
    - » Material differences
    - » Environmental differences
  - Require significant loss of material (e.g., x-rays)
  - Require delamination or blistering

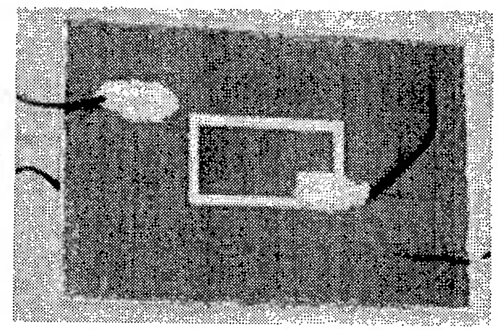
Several Sensor Versions are Available

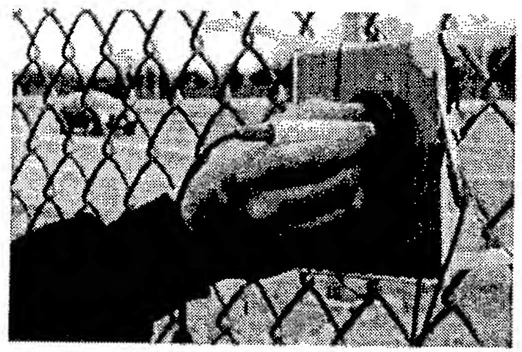
DSI

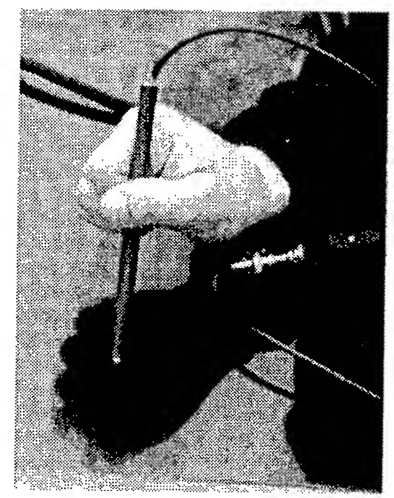
- Permanent and hand-held sensors meet different needs











Sensor Being Evaluated on Wide Range of Materials

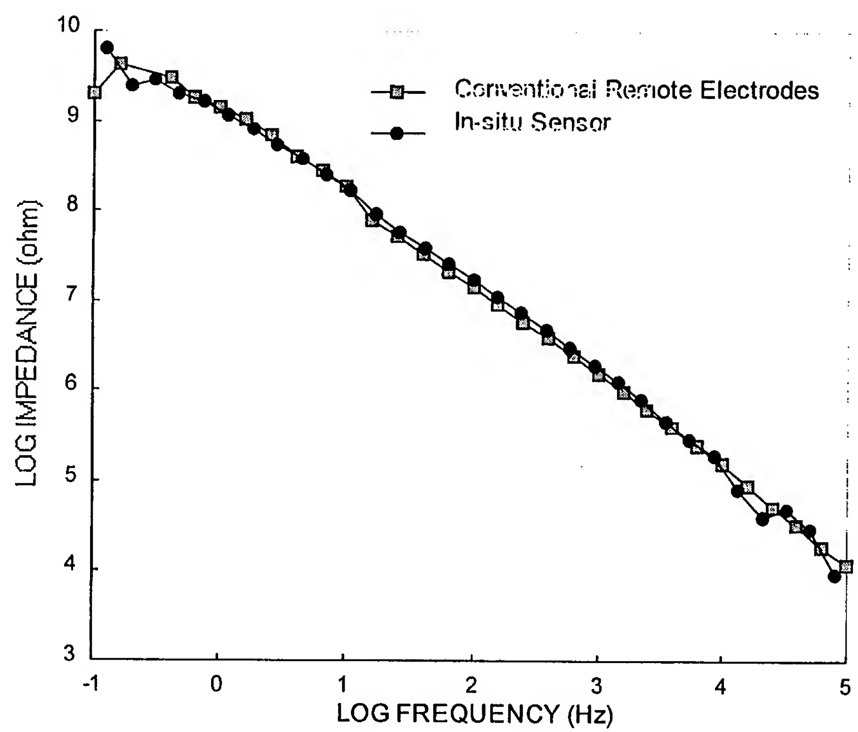
DSI

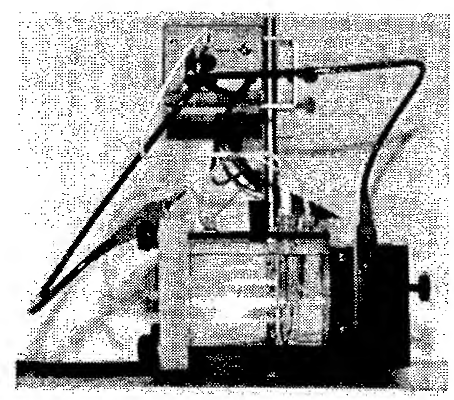
- Coatings
  - High performance coatings
    - » Film coatings
    - » Silicones
  - Aerospace/Navy coating systems
    - » Epoxy polyamide/urethane
    - » Waterborne epoxy/urethane
  - Automotive paints
    - » E-coatings
  - Commercial paints
    - » Alkyd
    - » Enamel
- Substrates
  - Cold rolled and galvanized steel
  - Aluminum
- Structures
  - Adhesive bonds
    - » Aluminum/epoxy
  - Composites
    - » Glass/polyimide
    - » Graphite/epoxy
- Environments
  - Salt fog
  - Humidity
  - SO_2
  - Immersion
  - Ambient field/lab

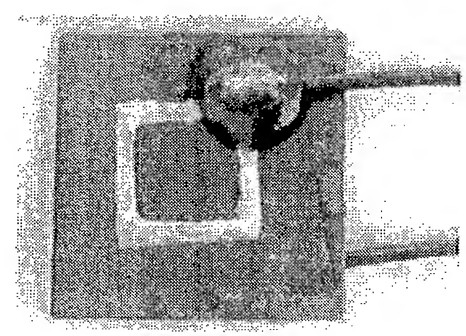
Permanent Sensor and Conventional EIS Give Identical Results

DSI

- *In-situ* sensor gives EIS results identical to those using conventional remote electrodes



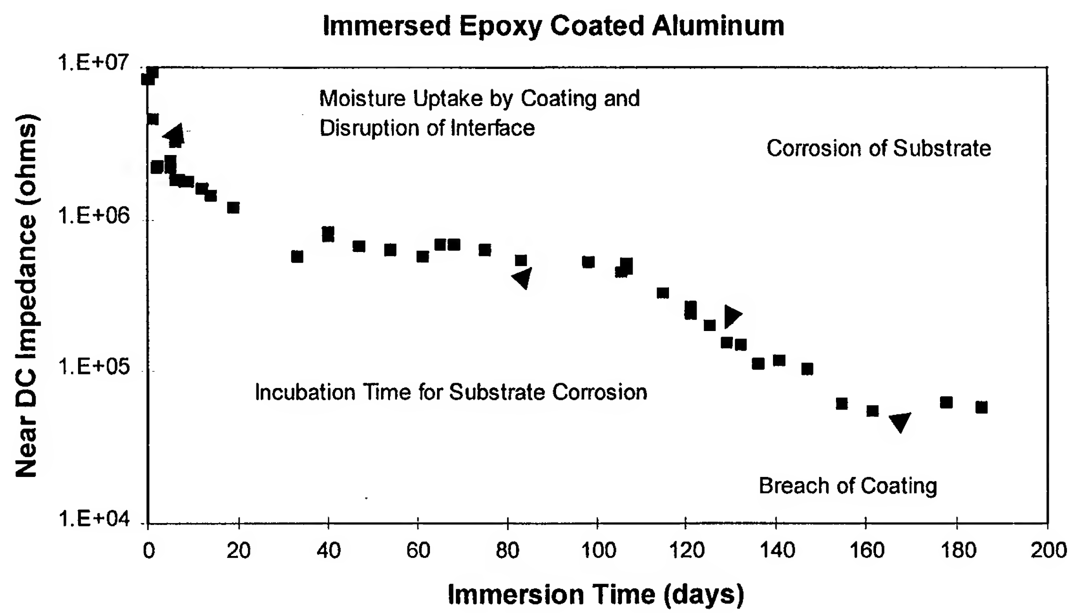




Sensor Detects Corrosion Stages

DSI

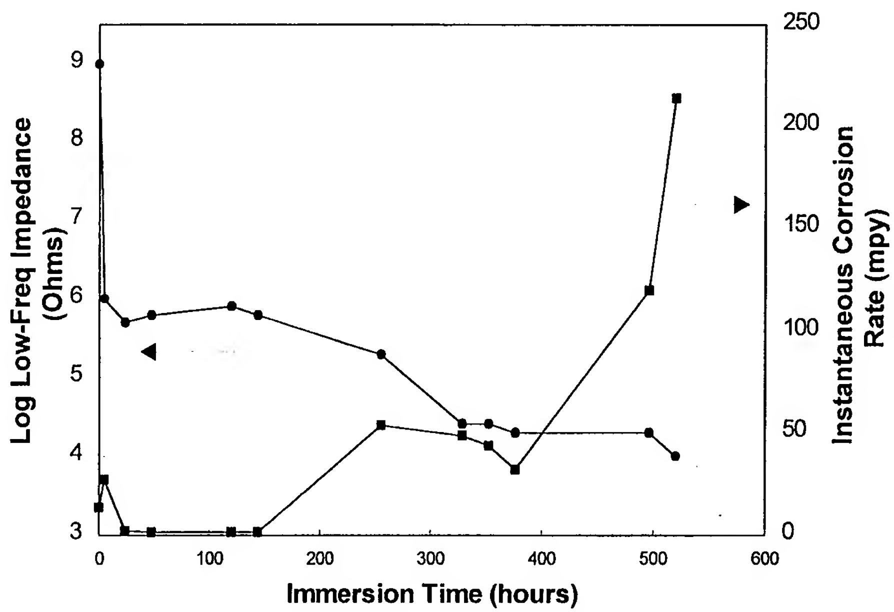
- Low-frequency impedance decreases in 4 stages
  - Very rapid change - moisture uptake
  - Stabilization/little change - corrosion incubation
  - Rapid change - substrate corrosion
  - Little change - breach of coating

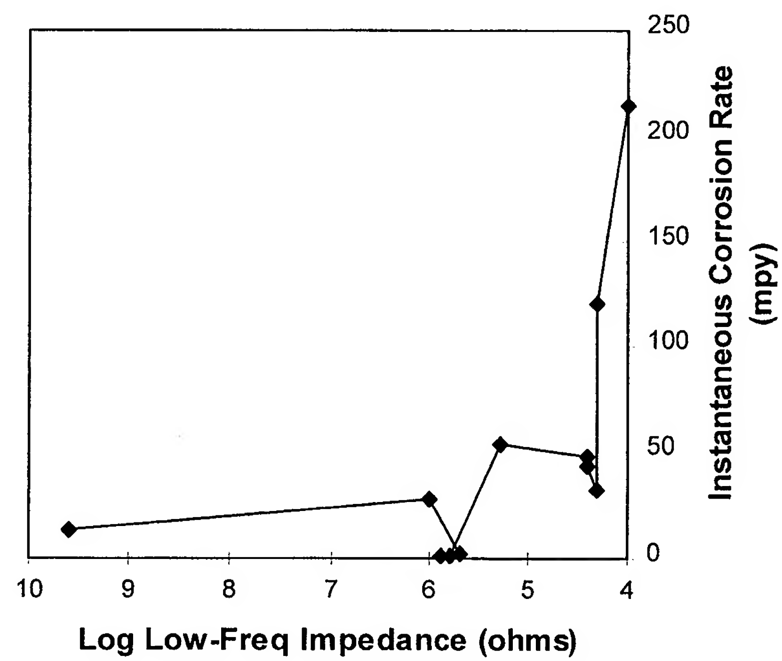


Sensor Results Correlate with Corrosion Rates

DSI

- EIS measurements are correlated with corrosion rates and ellipsometric data

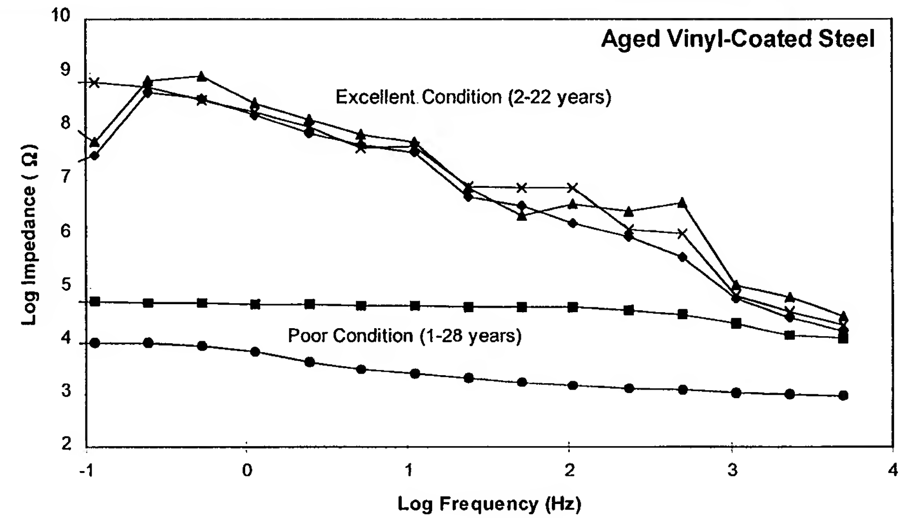


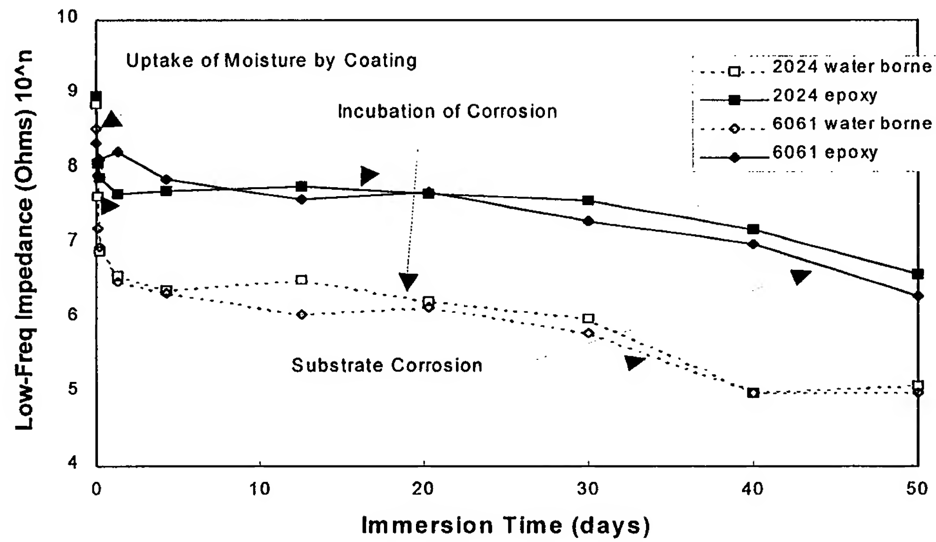


Sensor Measures Coating Effectiveness

DSI

- Differences in paint protectiveness readily monitored
  - Sensor results correlate with long-term performane of vinyl-coated steel
  - Waterborne coating is less effective than epoxy coating

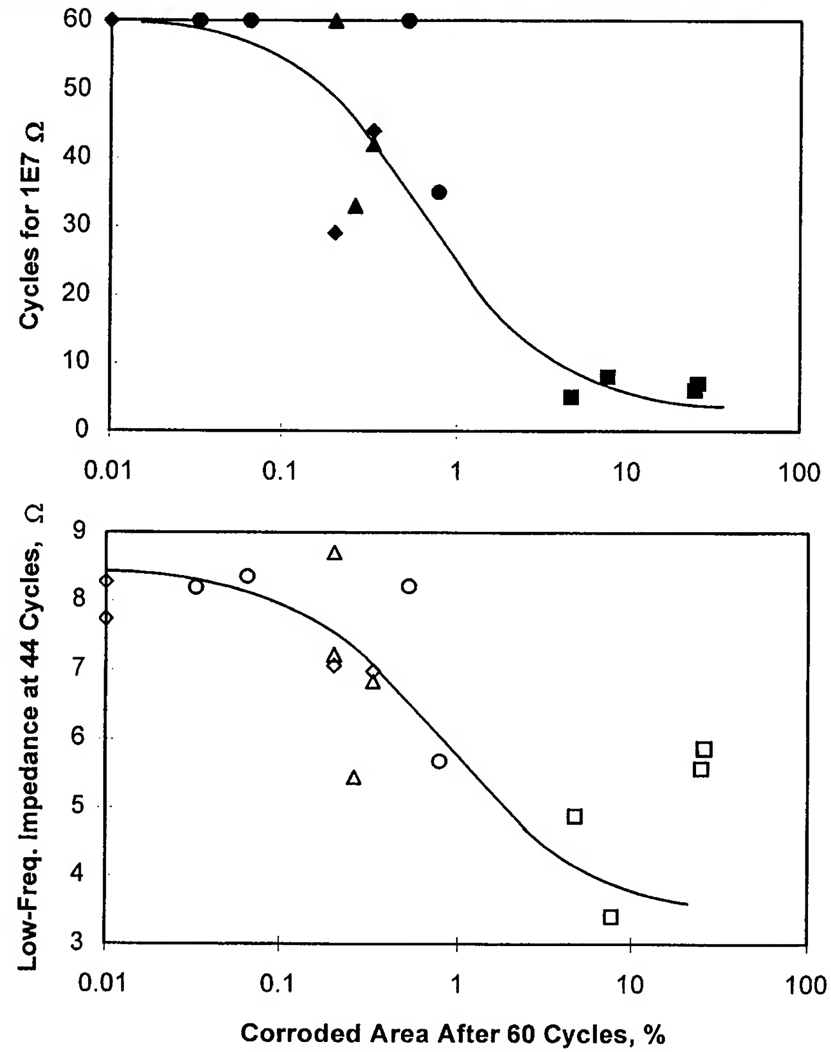




Sensor Predicts Extent of Corrosion

DSI

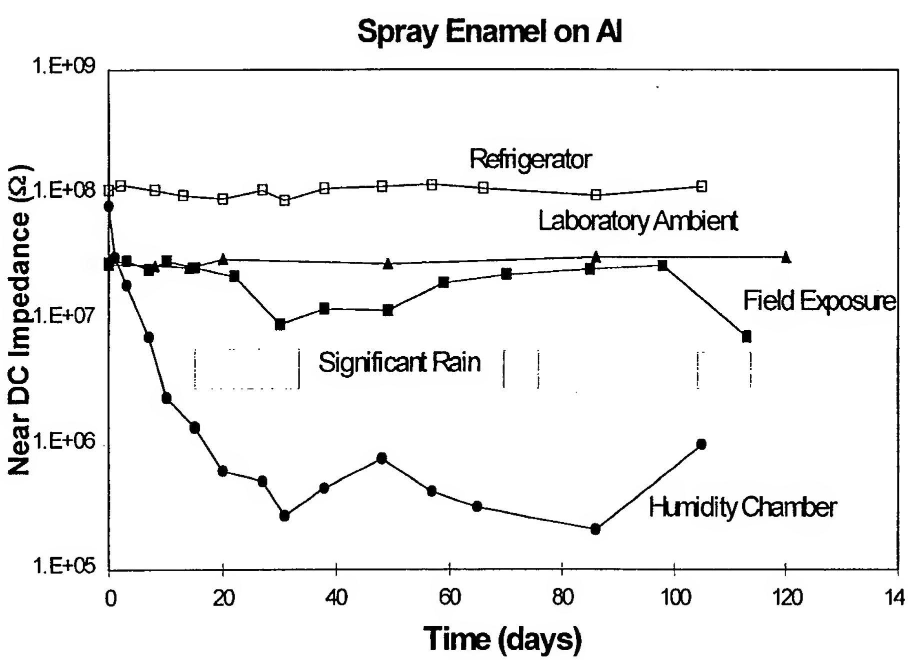
- In cyclic laboratory tests, sensor measurements correlate with future corrosion
  - Time for low-frequency impedance to reach $10^7\ \Omega$
  - Low-frequency impedance at an earlier time

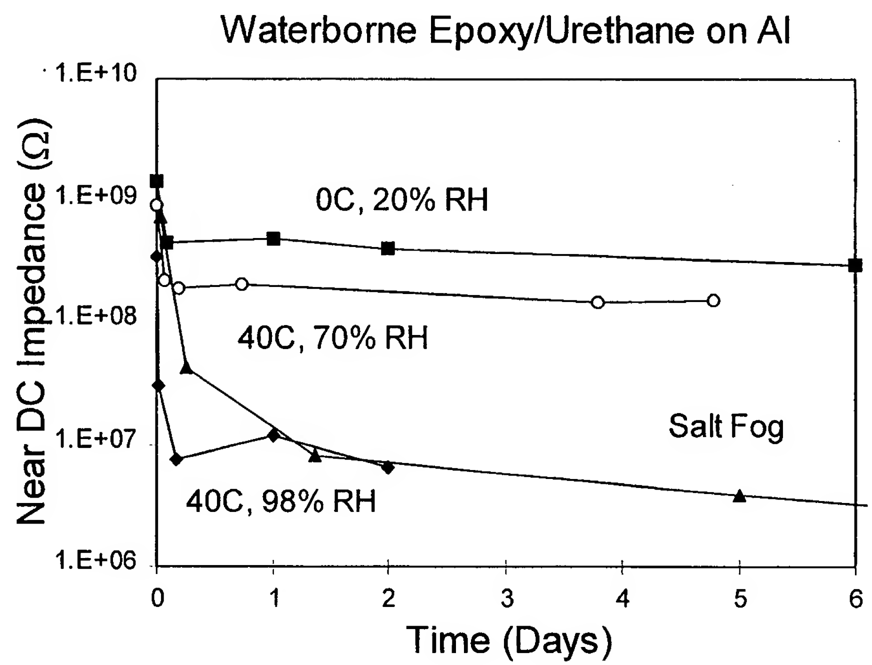


Sensor Shows Effects of Environment Severity

DSI

- Degradation of coatings occur at different rates in different environments

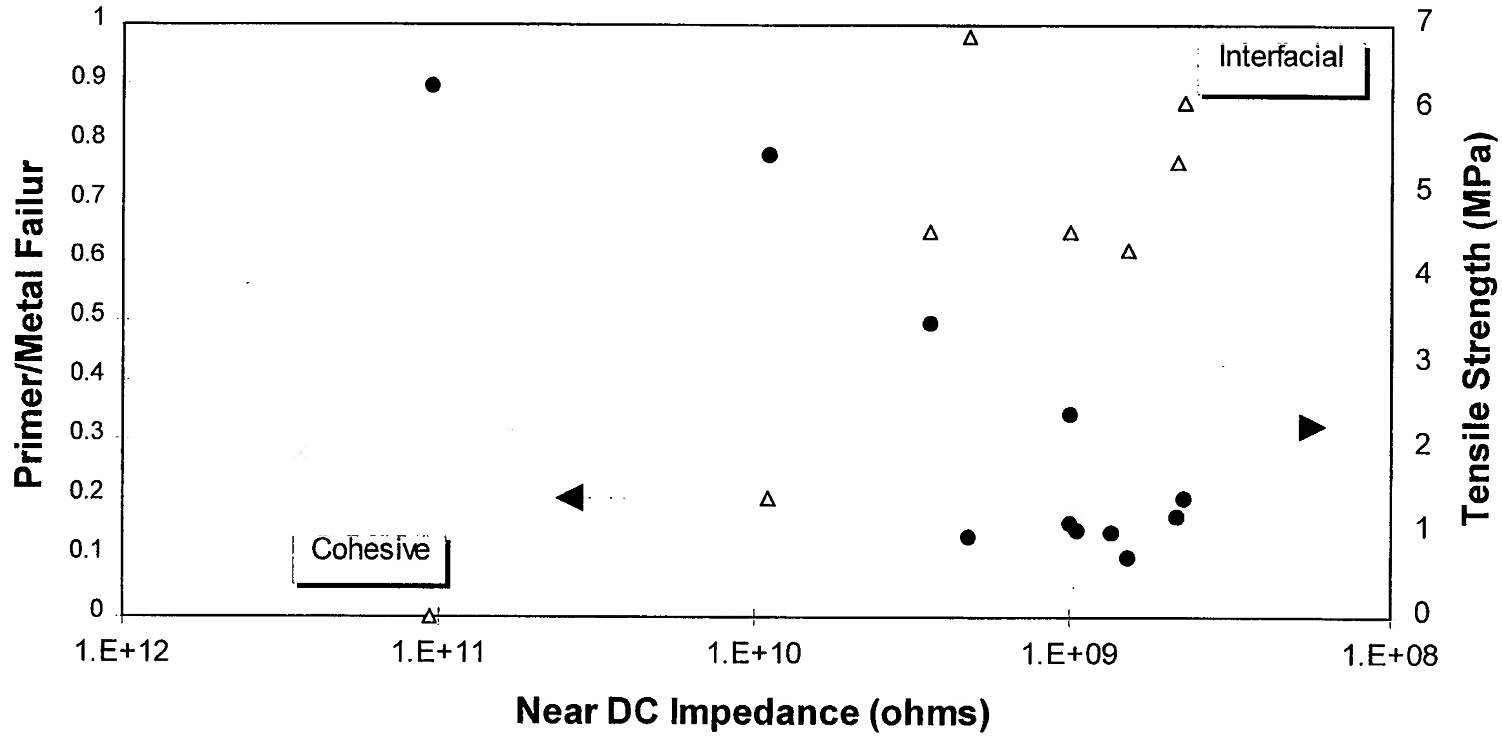




Paint Adhesion Correlates with Sensor Results

DSI

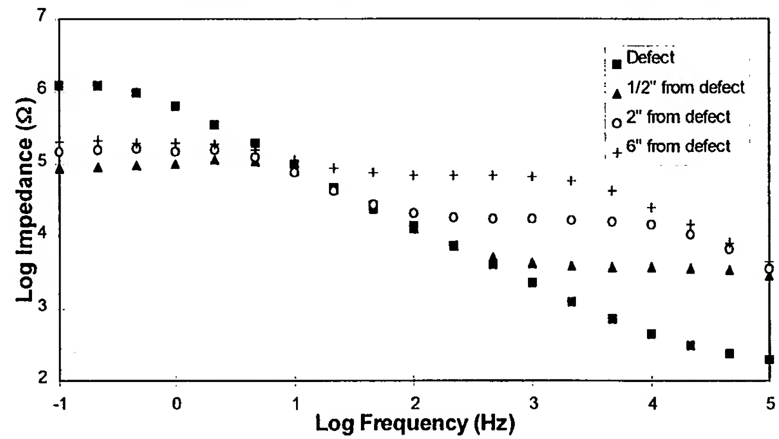
- EIS is correlated with delamination and adhesion of painted metals

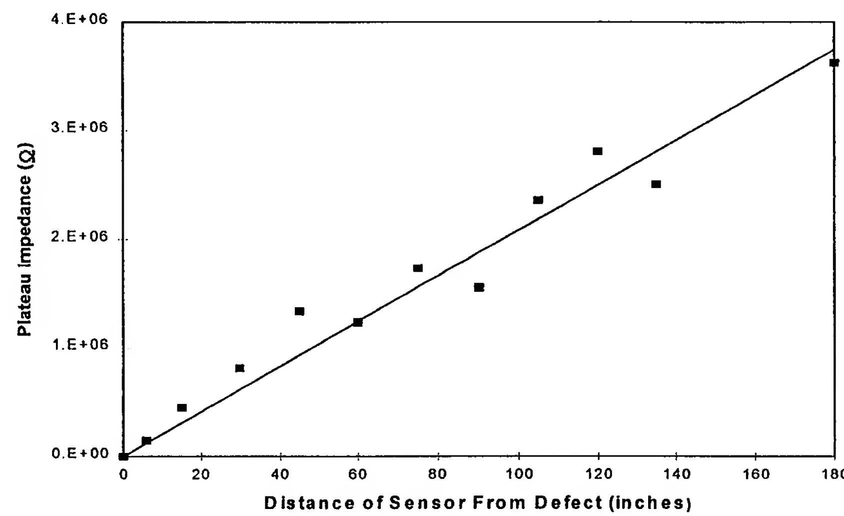


Defects Can Be Detected Away from Sensor

DSI

- Defects do not need to be at sensor
- Defects are detected up to 15 feet from in sensor in laboratory tests

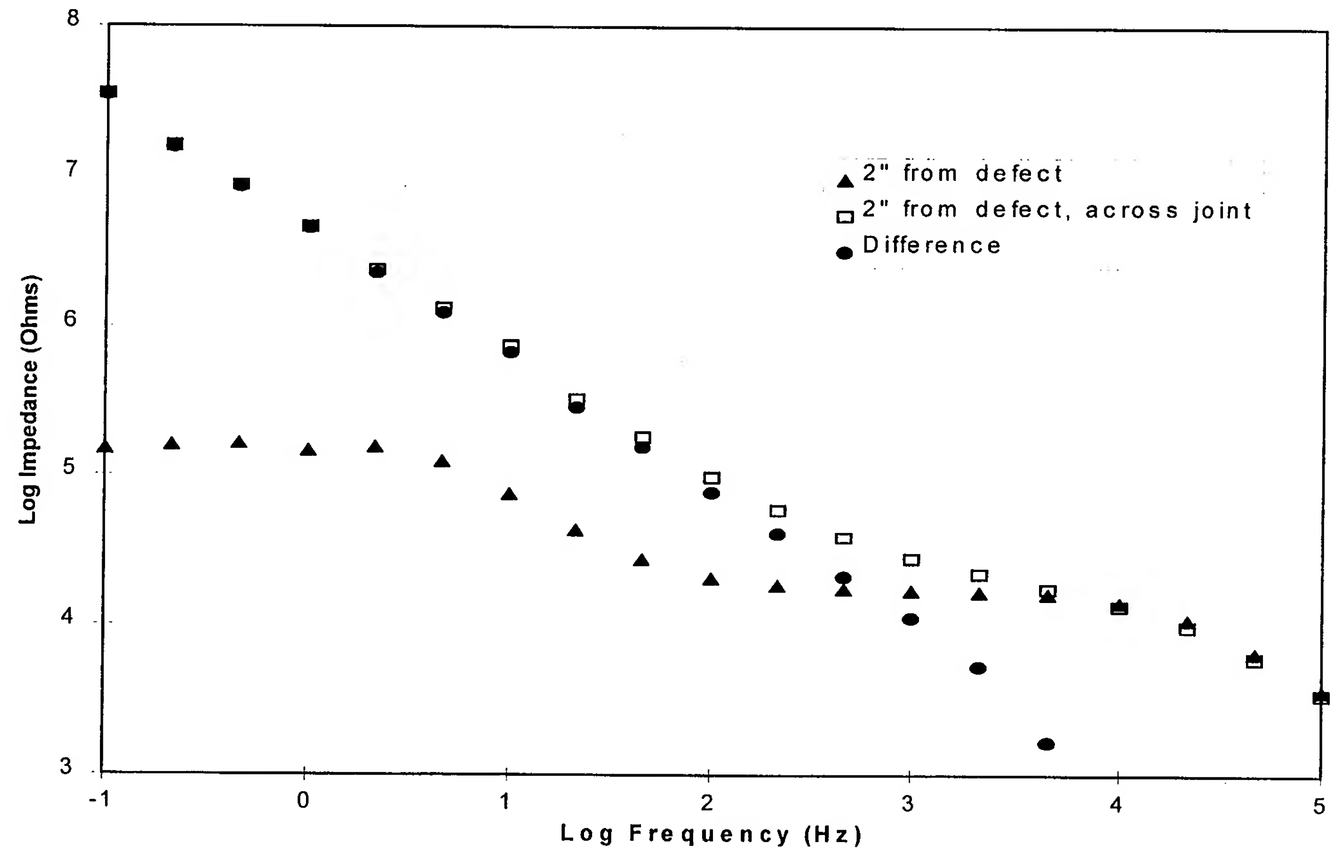




Sensor can also monitor across bonded joint

DSI

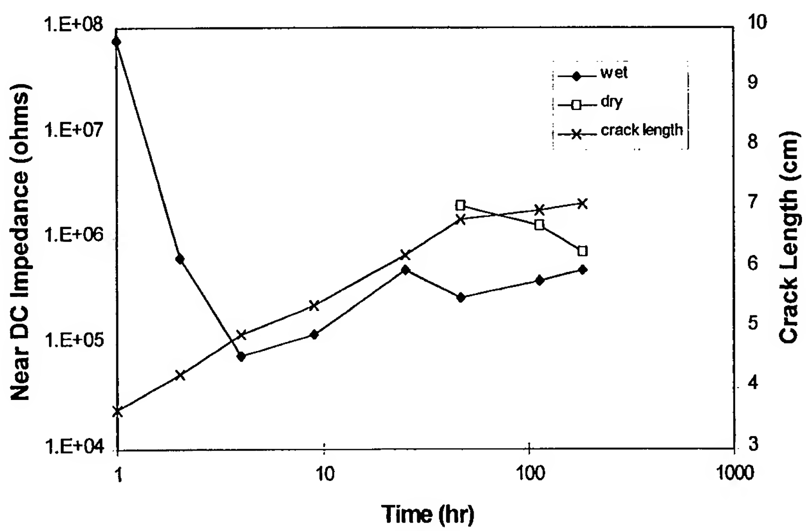
- Paint and adhesive are each inspectable

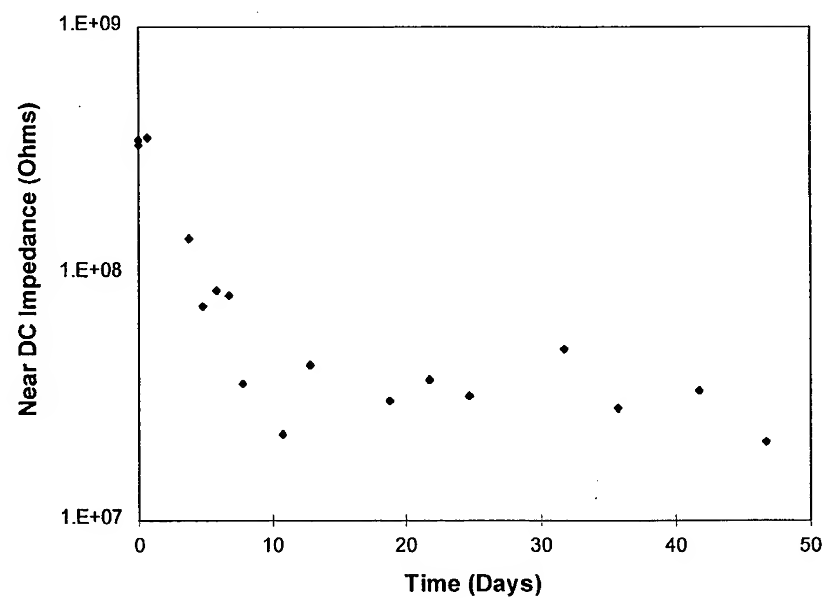


Sensor Detects Water in Adhesive Bondline

DSI

- Impedance is strongly dependent on the amount of moisture present in bondline



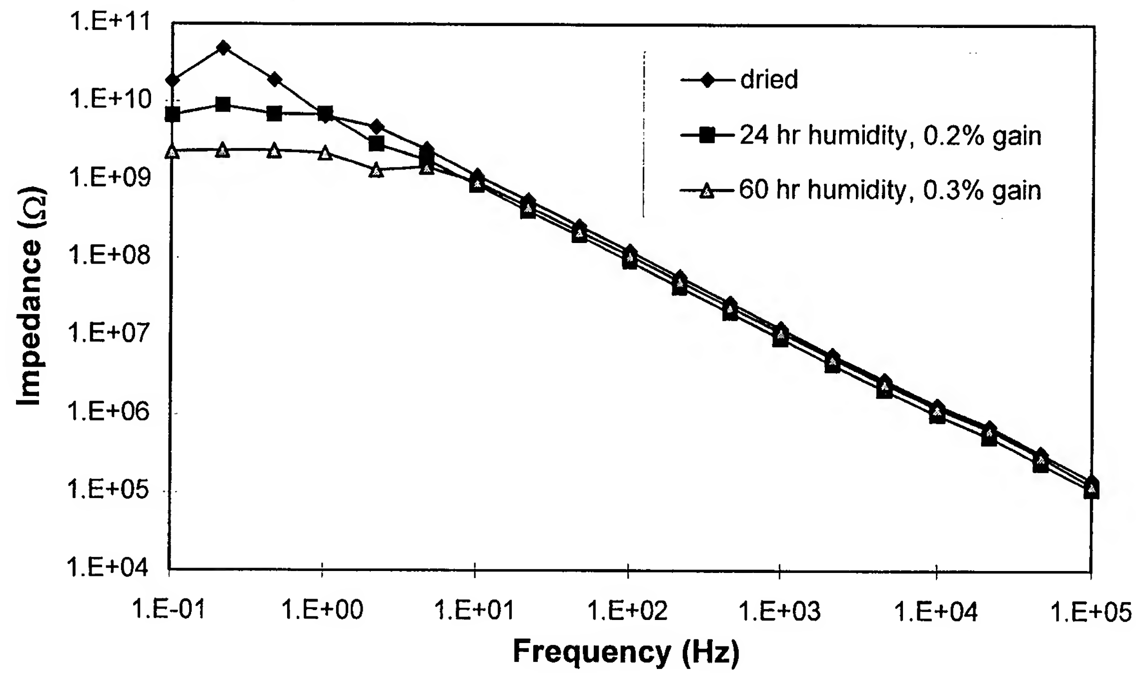


Sensor Detects Moisture in Composites

DSI

- Sensor detects low levels of moisture in glass/epoxy composites

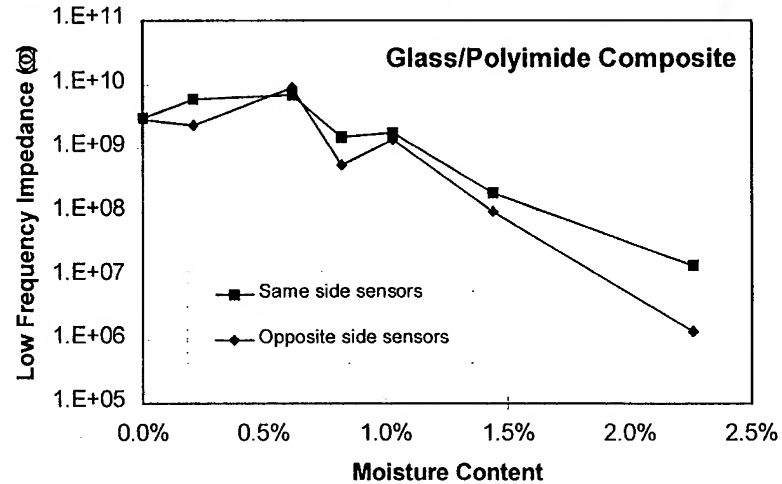
Fiberite 7701 w/ 7781 Weave E-glass,
8 Plies (63 mils)

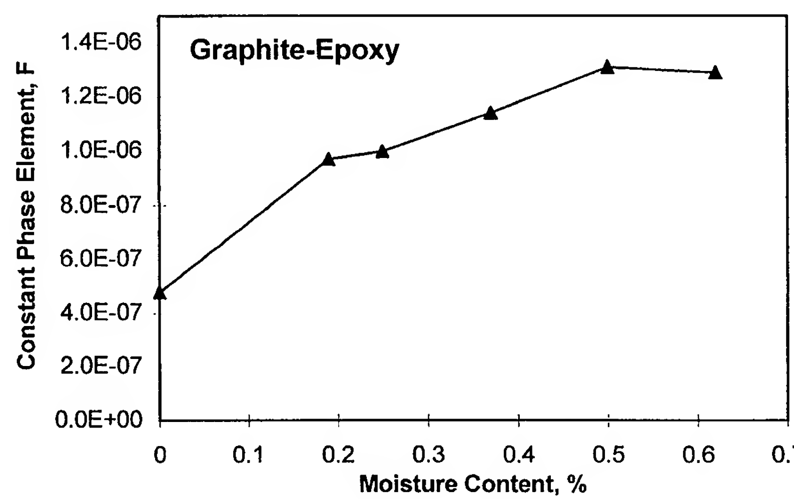


Sensor Detects Moisture in Composites

DSI

- Moisture absorption in composites can be detected using two sensors on a single side or on opposite sides
  - Glass-polyimide
  - Graphite-epoxy
- Equivalent circuit modeling helpful for graphite-epoxy





Conclusions

DSI

- Sensor gives equivalent measurements to conventional laboratory EIS
- Sensor detects coating degradation
  - Laboratory
  - Field
- Three stages of corrosion detected
- Permanently attached and portable, hand-held sensors give equivalent results

Conclusions (continued)

DSI

- Differences in coating effectiveness easily seen
- Measurements correlated with corrosion rate
- Coating defects detectable over wide area
- Moisture in composites detected
- Adhesive bond integrity also inspectable

DSI

Potential Uses of Sensor

- Coating Development
  - Correlate accelerated testing to proving ground testing to actual uses
  - Rapid screening of different coatings
- Critical Infrastructure/Equipment, Aircraft
  - Warn of impending coating failure
  - Needs-based maintenance

SESSION VI

FATIGUE AND CRACK GROWTH

Chairman - *R. Eastin*
Federal Aviation Administration

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An Operating Unit Of Mercer University

Probabilistic Stress Spectrum Generation (*ProSpectra*)

USAF ASIP Conference

San Antonio, Texas

December 3, 1997

Mary W. Schleider*
Robert A. Babb
Mercer Engineering Research Center
Warner Robins, Georgia

Randy Jansen
WR-ALC/TIED
Robins AFB, Georgia

121600-97016-F1

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OUTLINE

- BACKGROUND
- OBJECTIVE
- METHODOLOGY
- PROGRAM DESCRIPTION
- USER INTERFACE
- VALIDATION
- SUMMARY

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BACKGROUND

- Current Stress Spectra Generation for Weapon System Specific Missions
- Existing Tools "Hard-Wired"
- Often Program "Black Box" for User
- Existing Tools Limited in Generating Non-Standard Missions

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OBJECTIVE

- Provide Tool for Simulation-Based Methodology for Generation of Stress Spectra (Variables, Relationships, Sequence of Events)
- Provide Flexible Tool for Custom Tailoring
- Provide Option for Probabilistic Environments

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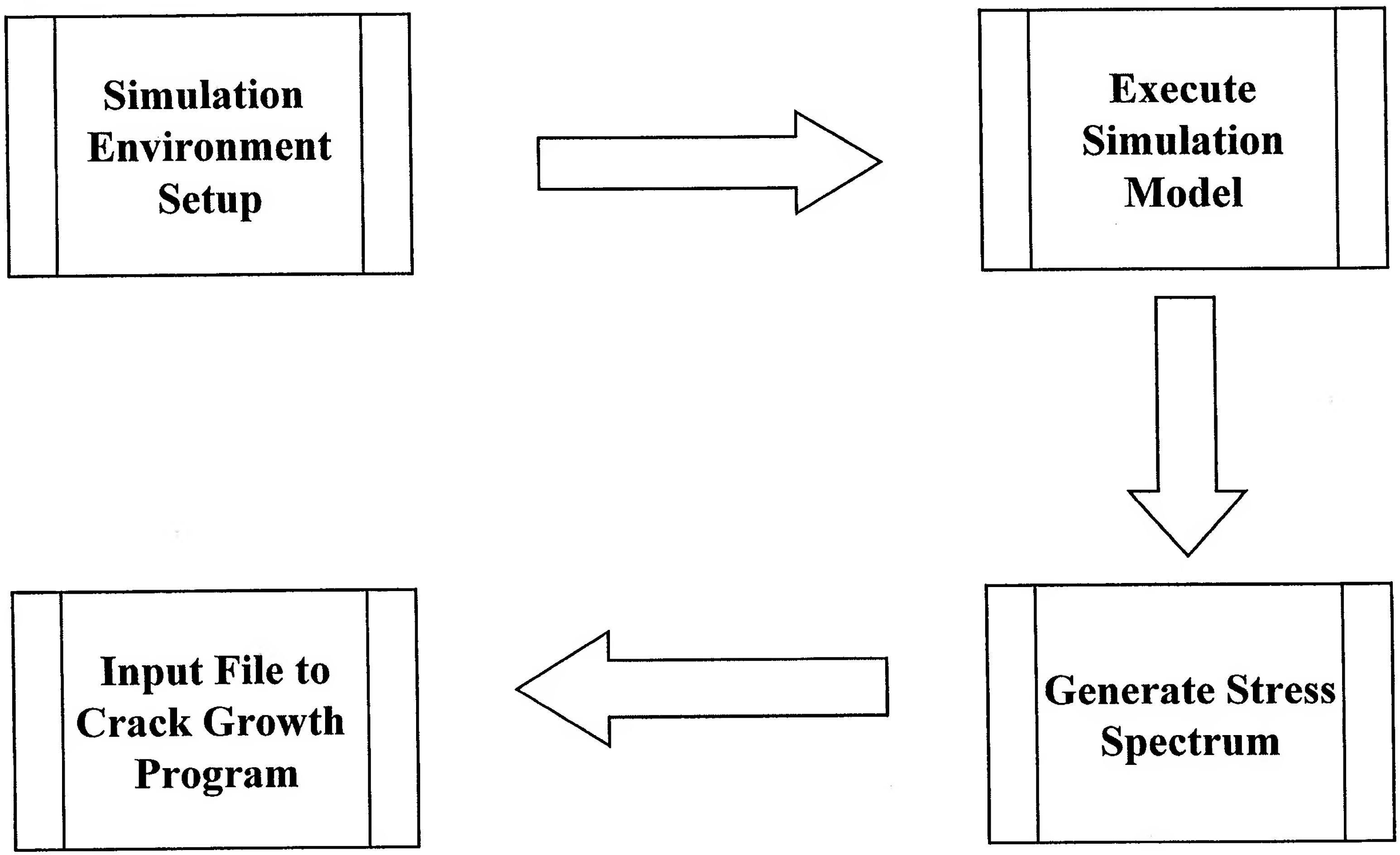
PROGRAM METHODOLOGY



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PROGRAM METHOLOLOGY

- ProSpectra Creates Stress Spectra based on Parameters, Events, Segments, Missions, and Mission Mixes
- Definitions:
  - Parameter- One of Many Conditions of the Environment
  - Event- Stress or Load, Discrete or Continuous
  - Segment- One or More Events
  - Mission- One or More Segments
  - Mix- One or More Missions

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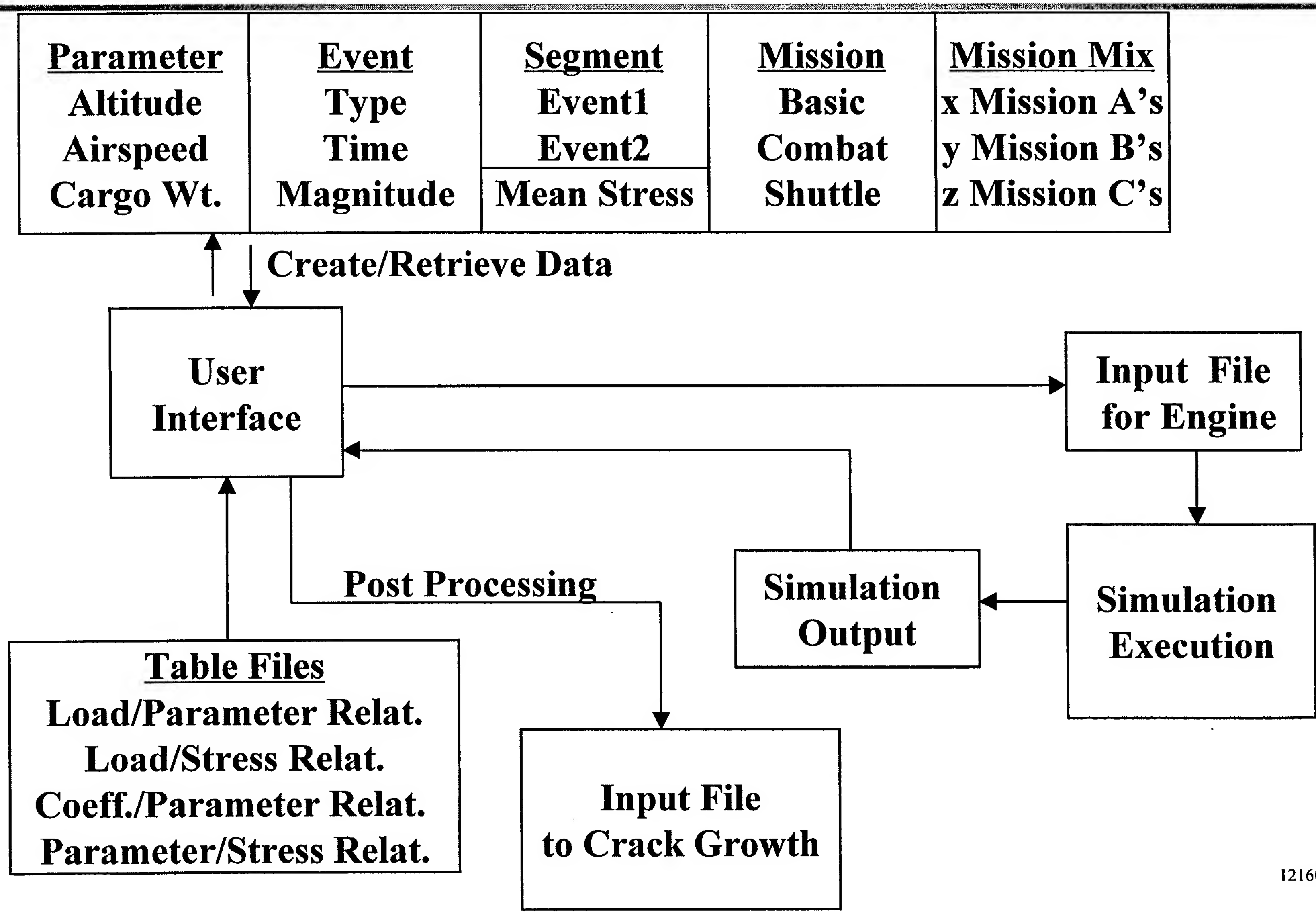
PROGRAM METHOLOLOGY



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PRE-PROCESSOR

- Simulation Set-Up
- Create/Modify Parameters, Events, Segments, Missions, and Mission Mixes
- Table File Lookups for Values, Relationships
- Options of Random Magnitude, Order, Timing, Distribution

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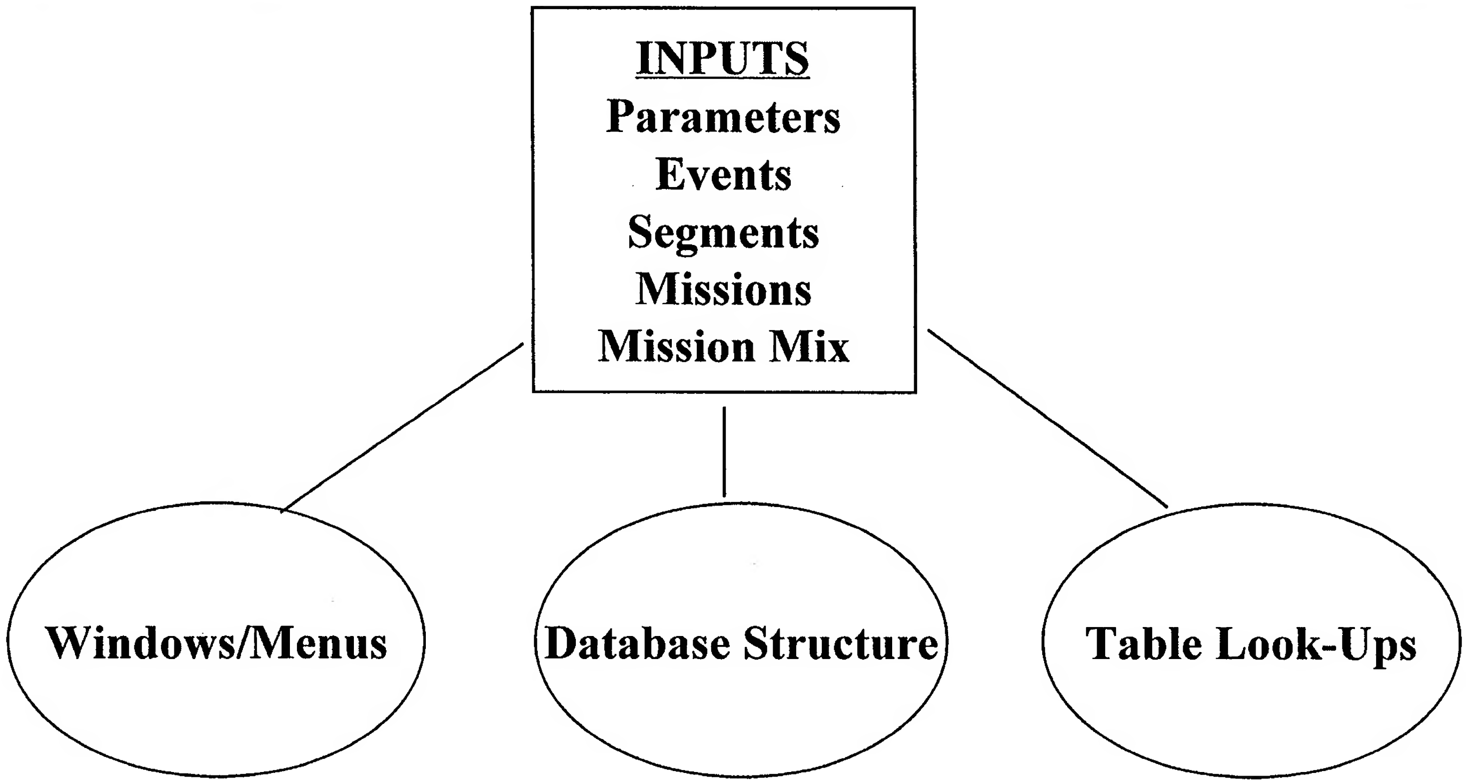
PRE-PROCESSOR



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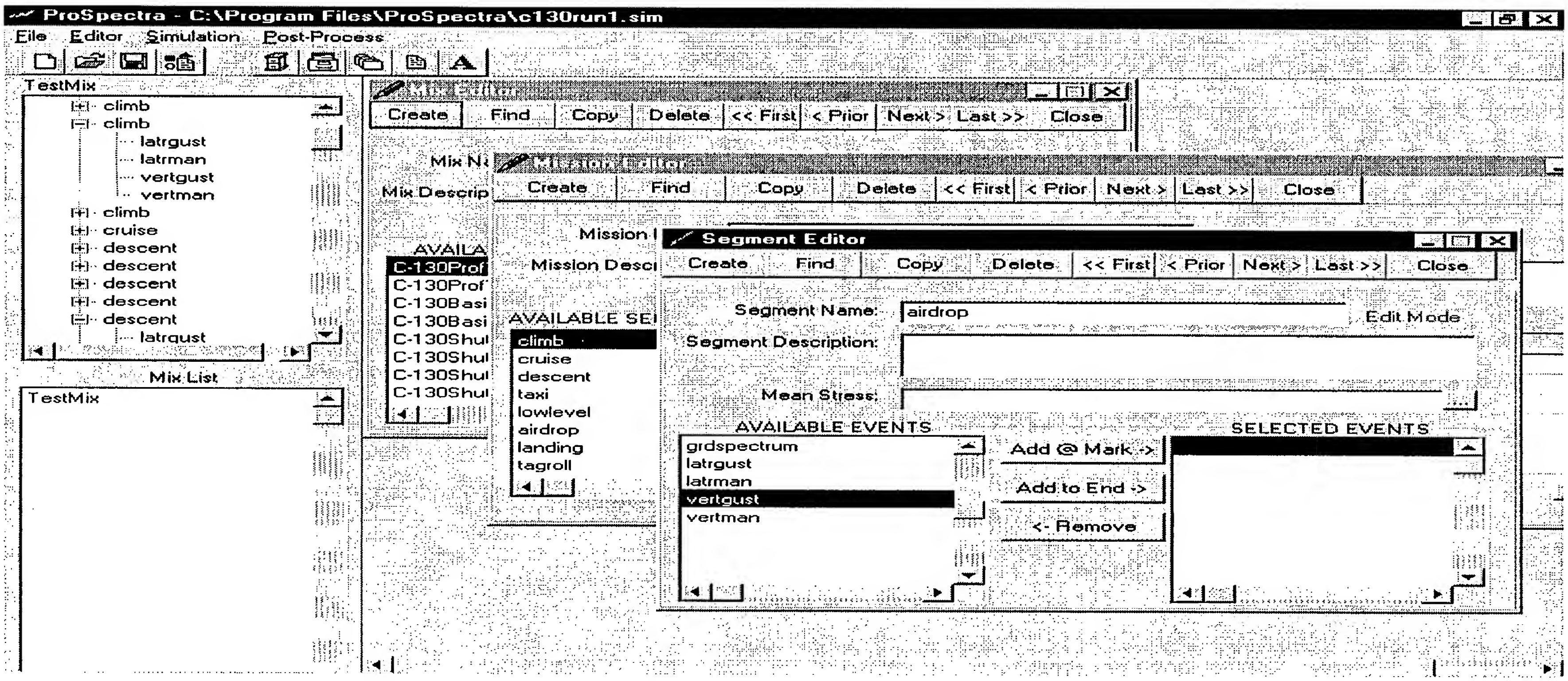
PRE-PROCESSOR



TYPICAL PRE-PROCESSOR SCREEN

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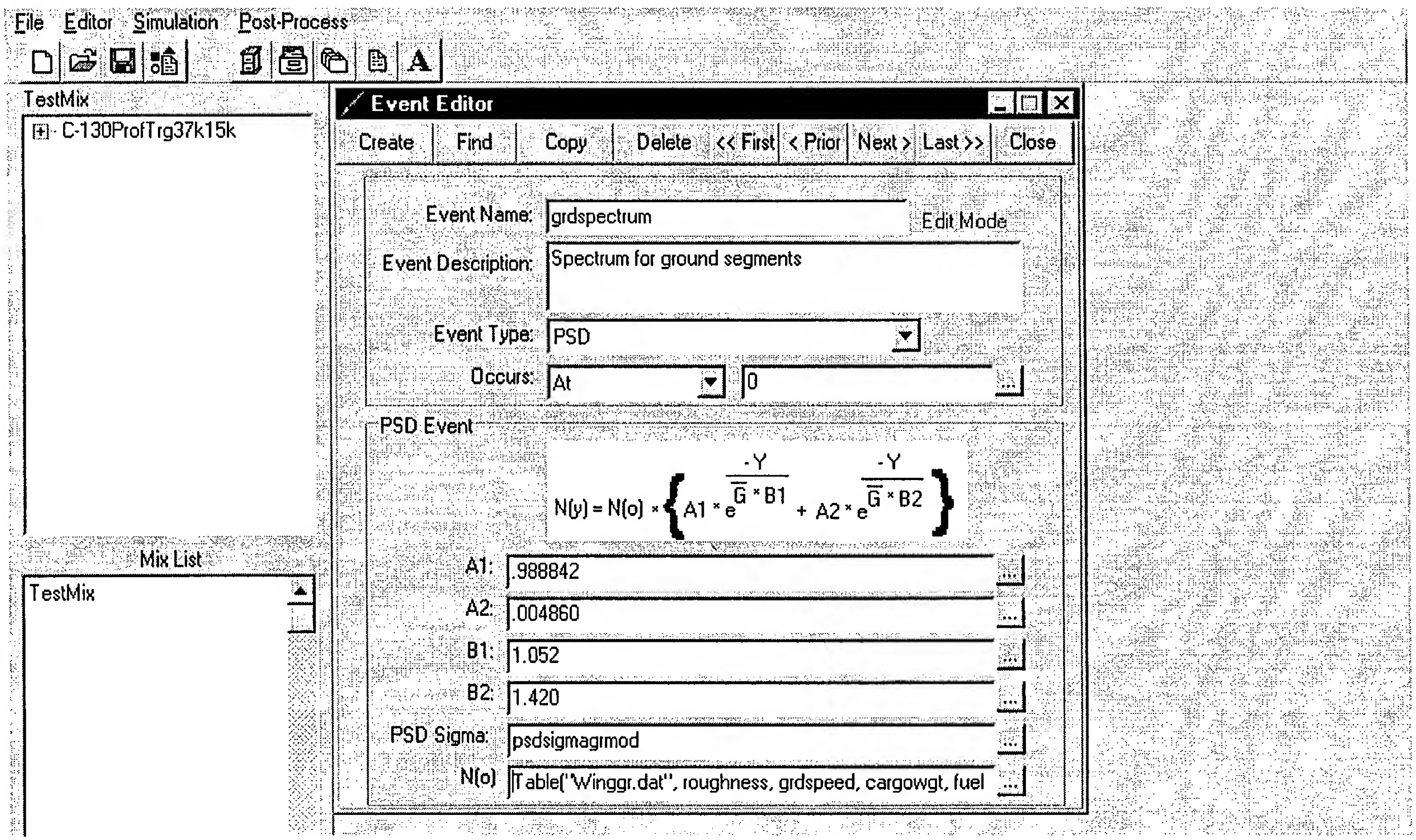
PRE-PROCESSOR



EVENTS EDITOR WINDOW

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PRE-PROCESSOR

Altitude	Airspeed	Cargo Wt.	Fuel Wt.	Bend.Mom.
500.00	125.00	0.00	2300.00	2800000.00
500.00	125.00	0.00	17000.00	1800000.00
500.00	125.00	0.00	32000.00	700000.00
500.00	125.00	0.00	45000.00	1900000.00
500.00	125.00	10000.00	2300.00	2100000.00
500.00	125.00	10000.00	17000.00	1600000.00
500.00	125.00	10000.00	32000.00	600000.00
500.00	125.00	10000.00	45000.00	2401000.00

Bend.Mom.	Mean Stress
-1100000.00	-35000.00
0.00	0.00
1100000.00	35000.00

SAMPLE TABLE FILES

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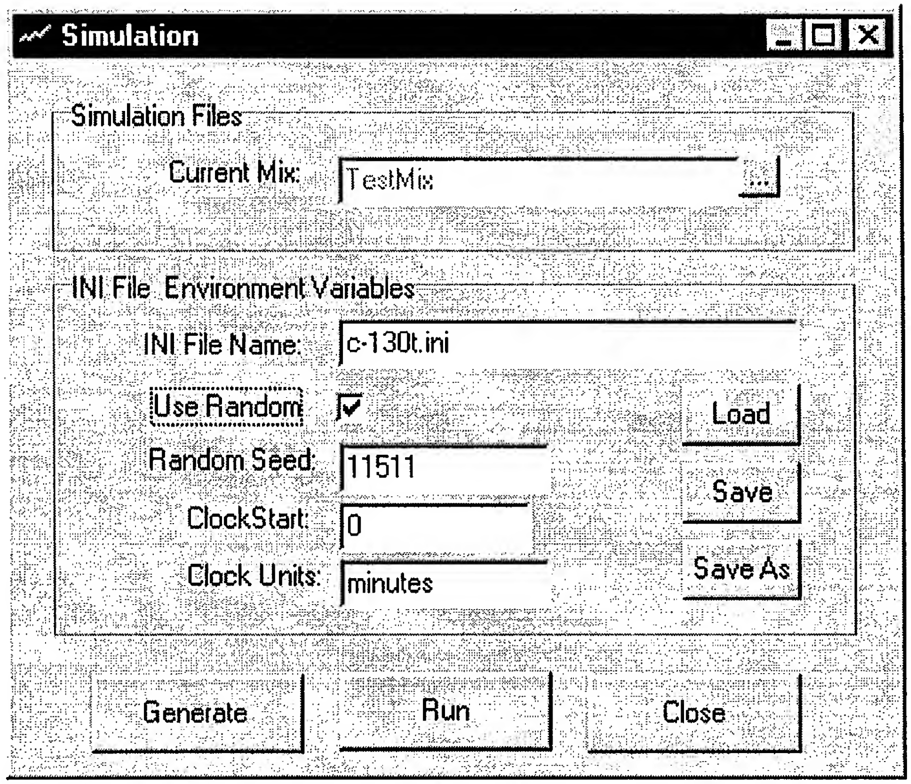
PRE-PROCESSOR



SETUP FILE WINDOW

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PROCESSOR

- Simulation Based on User-Defined Mission Mix
- Generates Times and Numbers of Events (Random, Function, Table Look-Up)
- Output of Discrete, PSD, and Constant Amplitude Stresses

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POST-PROCESSOR

- Stress Spectrum Generation
- Cycle Counting, Ordering, and Layering
- Graphical Displays

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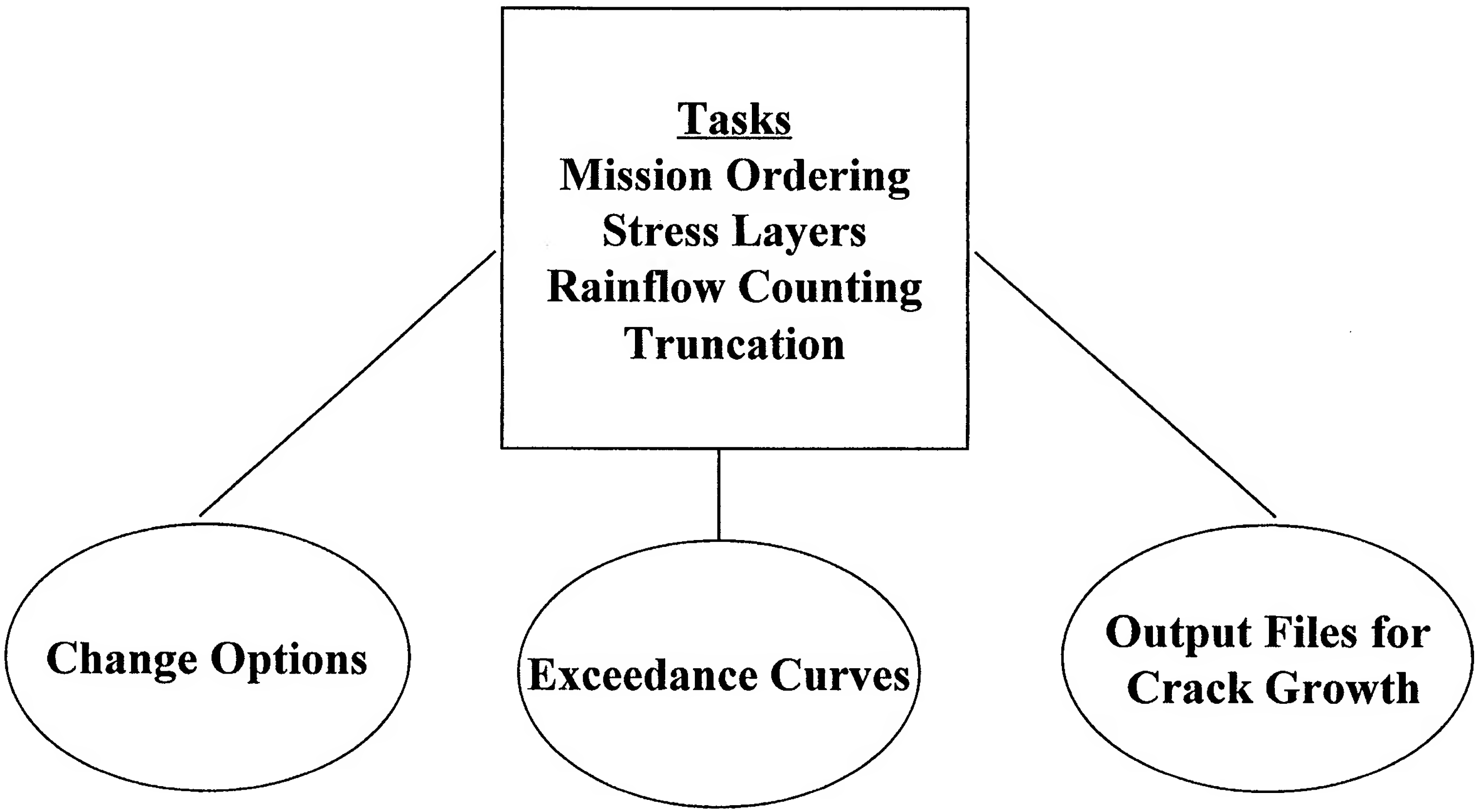
POST-PROCESSOR



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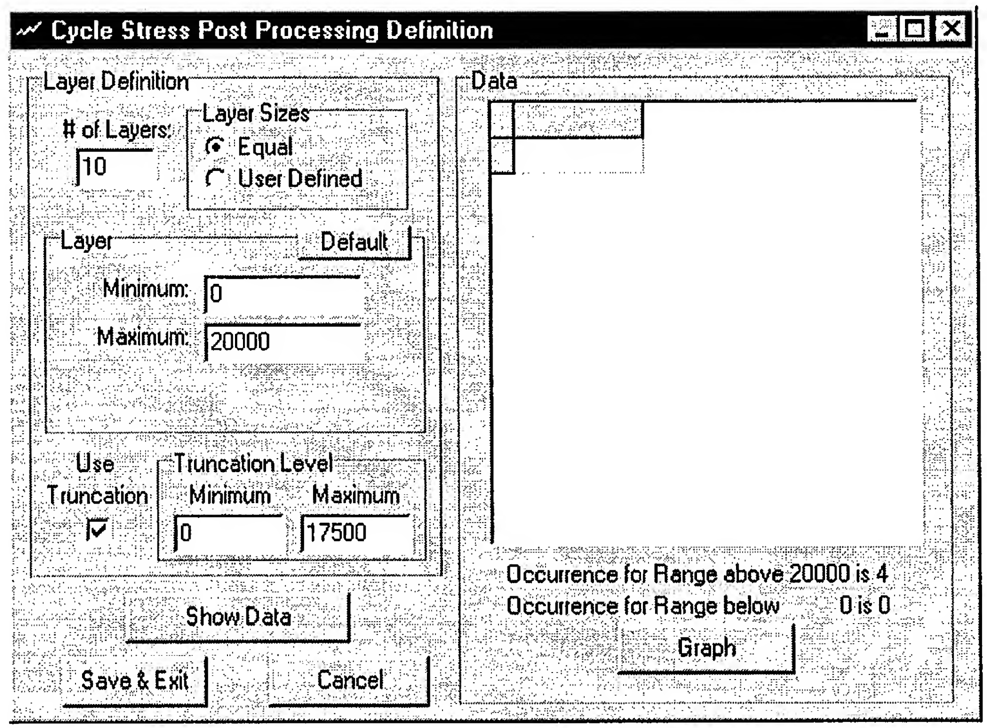
POST-PROCESSOR



TYPICAL POST-PROCESSING SCREEN

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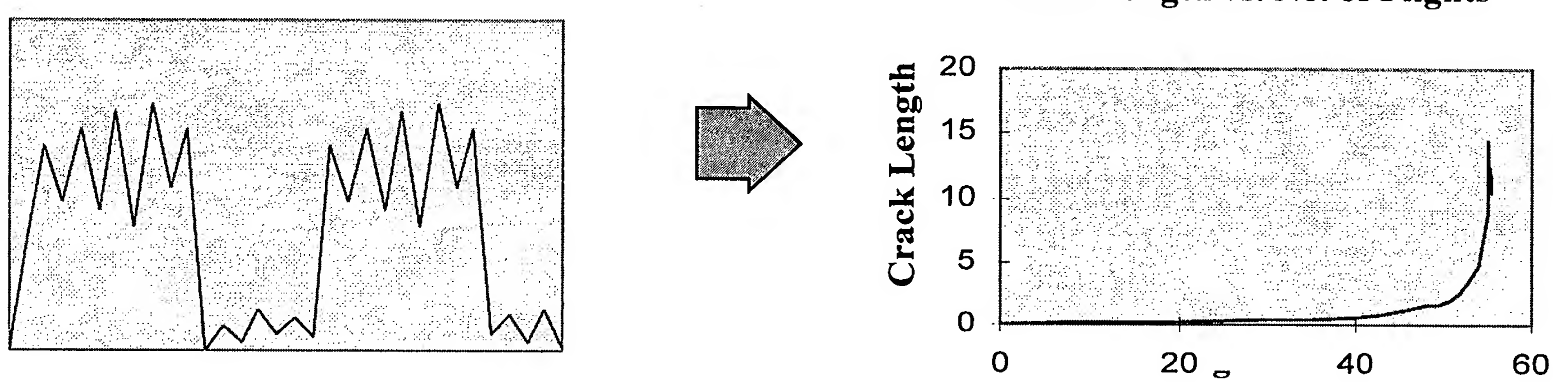
VALIDATION

Comparison Between Crack Growth Prediction from ProSpectra Stress Spectrum and Existing Deterministic Model Spectrum

Stress Spectrum

Crack Length vs. No. of Flights



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SUMMARY

- *ProSpectra* is a Windows™-Based Tool for User-Generated Stress Spectra
- Probabilistic and Deterministic Events Simulated
- PSD, Constant Amplitude Cycles, and Discrete Events Processed
- Development of Mission Profiles Possible from New and Limited Data

121600-97016-F1

Damage Accumulation and Spectrum Editing

Eric J Tuegel and Craig L Brooks

Analytical Processes / Engineered Solutions
APES, Inc.
3542 Oxford Ave. St. Louis, MO 63143
Phone: (314) 644-6040

1997 USAF Structural Integrity Program Conference

The application of durability and damage tolerance analysis tools to aircraft structure is enhanced by having a better understanding of the contents and significance of damage in variable amplitude spectrum.

The intent of this presentation is to describe the advantages offered by damage tables in understanding the contributions of each element of a spectrum to the life predictions.

Significant benefits can be achieved by utilizing simple concepts to edit spectrum to save test costs and improve processing time, obtain a better understanding of the life prediction process, and continue improvements in the methodologies.

APES, Inc. would like to acknowledge and thank the National Research Council of Canada for providing spectra and scenarios to exercise the concept on and demonstrate the practical application to a specified problem.

Fatigue Analysis is More than Just a Life

- **Occurrence and Damage Tables provide insight into the loading spectrum so many fatigue issues can be addressed more intelligently.**
  - Editing spectra for full-scale fatigue tests
  - Selecting proper materials
  - Identifying analyses limitations to establish confidence level
- **Database programs make this type of information available.**
  - Queries are written to interrogate the large tables that result
  - Forms are developed to report results in a useful format.

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Three advantages in using occurrence and damage tables to understand spectrum characteristics are:

1) Efficient editing of large spectra can be conducted to reduce the number of applied cycles for full-scale fatigue tests. Spectra for aircraft often need to include buffet and vibratory cycles along with the maneuver loads. Including all cycles into the test can result in very long tests with significant cost. Cost and time considerations require the elimination of many of the non-damaging cycles and load excursions.

2) Selection of proper materials for critical structure can be aided by the information in damage tables. Each material has unique attributes and behaviors. Comparing a material's strengths and weaknesses with the characteristics of the spectrum enable an optimum material selection for an intended application.

3) The limitations of analyses can be easily ascertained. Damage tables provide a means for determining the amount of damage contributed by load cycles which are outside the bounds of available data or of current theories.

The creation of occurrence and damage tables can generate a large amount of data and information. Coupling a fatigue analysis program with a database tool allows efficient access to the information.

Creating Occurrence and Damage Tables

- Initial Processing of a Spectrum
  - Normalization to the largest Peak or Valley
  - Bucket Selection
- Format of Tables
  - PEAK versus VALLEY
  - RANGE versus PEAK or VALLEY
- Interior of Tables
  - Occurrences
  - Damage measure
    - Crack Initiation Analysis: fraction of Miner's Rule contributed by cycles
    - Crack Growth Analysis: fraction of crack extension in a single spectrum pass at some a_o

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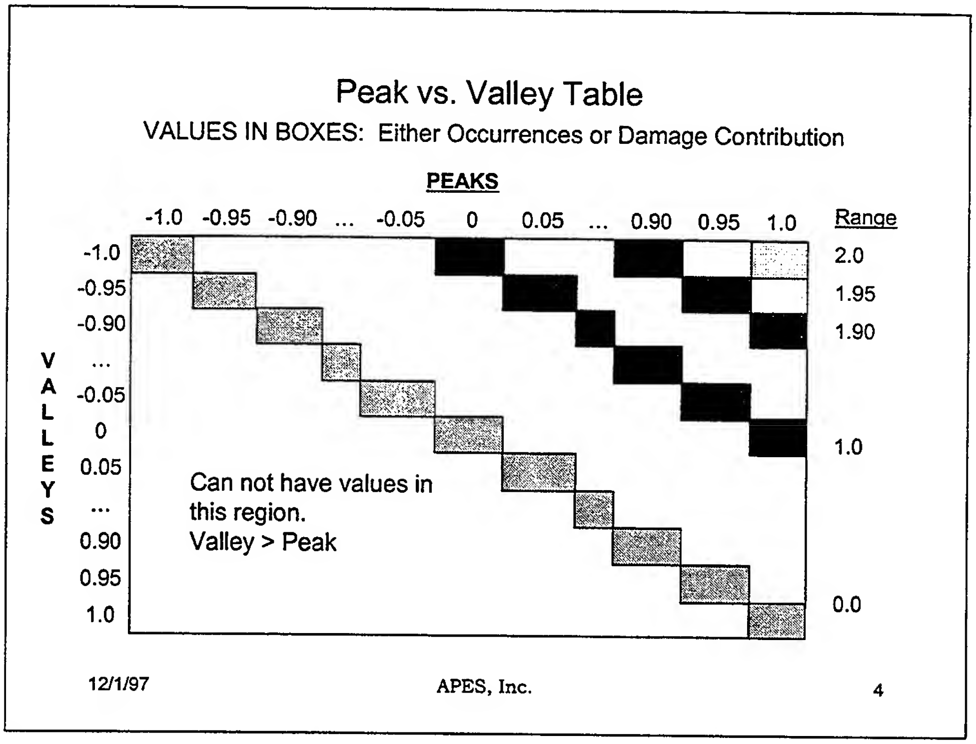
The spectrum load levels are first normalized to the largest magnitude load level in the entire spectrum, either the peak or valley whichever is greater. This creates a spectrum limited to values between 1 and -1. Often there are a large number of different peak-valley combinations in the spectrum. Each load level often carries several significant digits. Some sort of binning (or bucket) process must be used to group the cycles in order to keep the tables tractable. The normalized load levels are placed into defined incremental buckets. This creates a spectrum with a definable matrix of load levels and combinations. The binning can be done within the fatigue analysis program if desired. Or the spectrum can be modified prior to performing the analysis.

At APES, we have chosen to modify the spectrum by rounding off each spectrum turning point (peak or valley) to the nearest 0.05. This is typical engineering accuracy, however, smaller values such as 0.02 or 0.01 can be used in situations where there is a high degree of accuracy. A 0.05 increment results in 40x40 matrix.

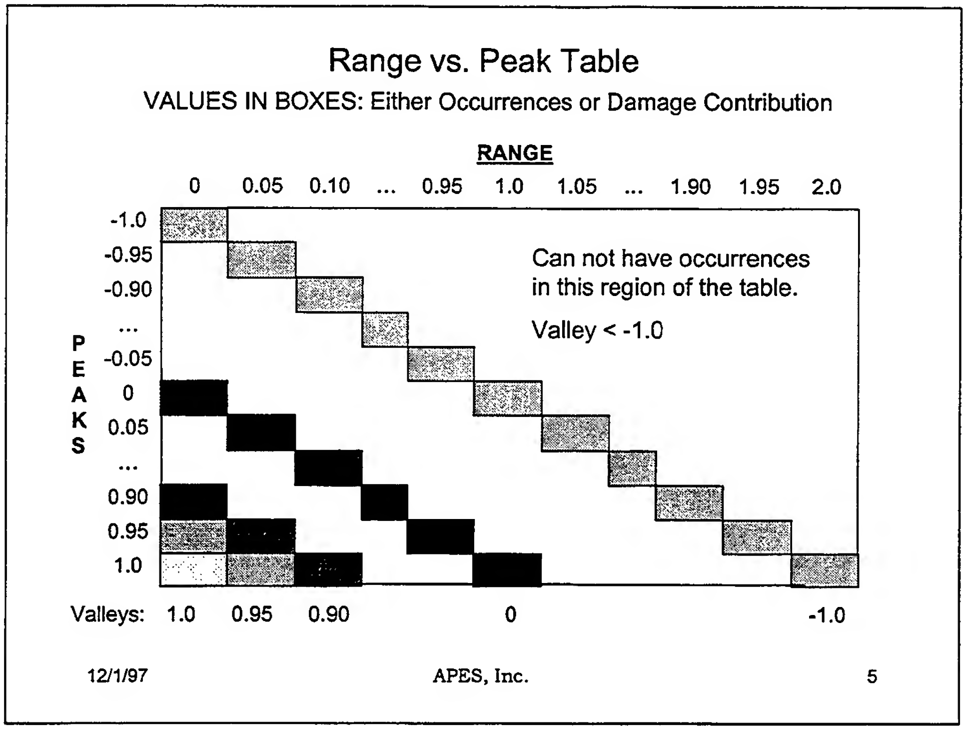
The interior of the tables provide the information needed to understand the contents and significance of the spectrum. The individual attributes of the damage tables will be discussed in subsequent charts.



This is an example of a table plotting peak vs. valley for the normalized spectrum. The increment size of each bucket is 0.05. The matrix captures every value within the normalized spectrum. The matrix is an upper triangular matrix. There should be no values in the lower left corner as this would imply a valley greater than a peak. The value to be reported in each peak-valley cycle box can either be the number of occurrences of that combination in the spectrum or the damage contribution of the combination.

The distribution of peaks and related valleys can readily by seen, either in terms of numbers of events or damage per group. In addition, by being familiar with the the patterns associated with the table one can recognize the contours of constant range, peak minus valley, as highlighted above. The diagonal is the contour of load cycles with zero range; the peak is equal to the valley.

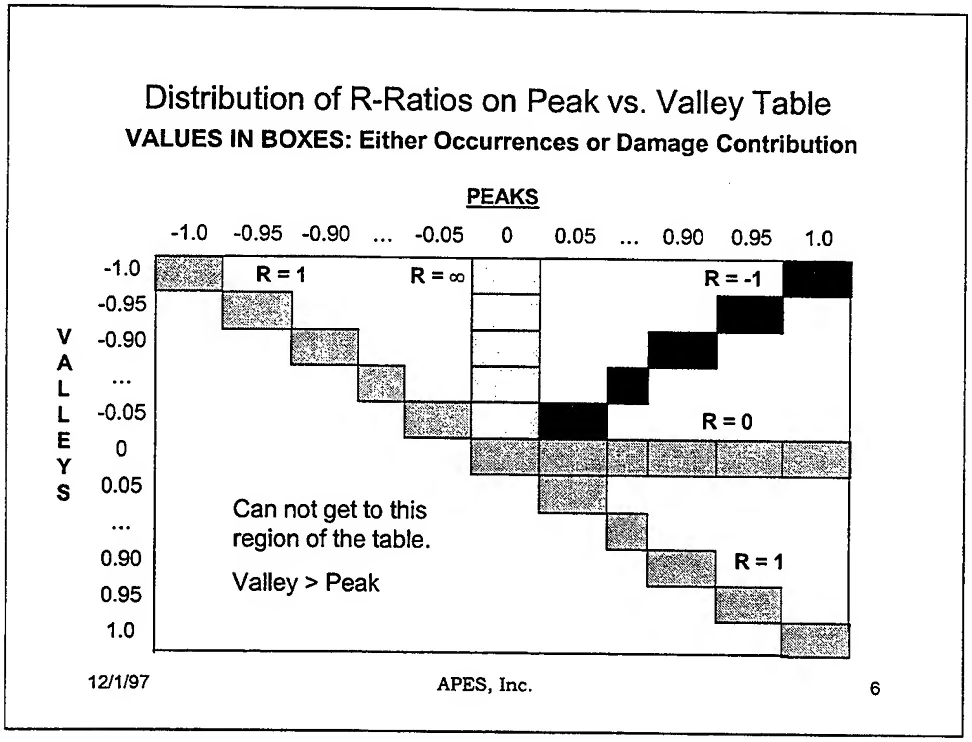
The original values in the spectrum can be recovered by multiplying the normalized values along the axes by the reference load or stress.



Another way to present the information is range vs. peak. For a spectrum normalized between 1.0 and -1.0, this matrix is a lower triangular matrix. Values in the upper right corner would require valleys less than -1.0.

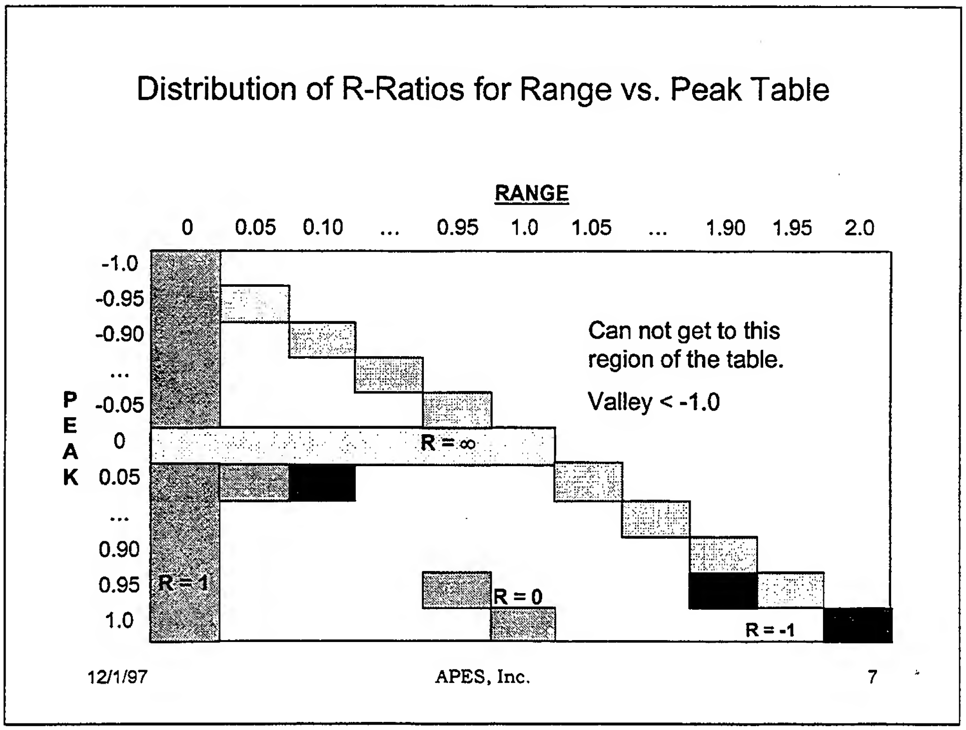
The concentrations of occurrences and damage provide information as to the major fatigue drivers in the spectrum. This format is especially useful for spectrum editing. The less damaging cycles are in the upper left corner; smaller range and lower peak stress. The cycles become more damaging as you move down and to the right. Thus cycles nearest the upper left corner are candidates for elimination.

For comparison with the previous table, contours of constant value for the valley are highlighted.



In addition to contours of constant range, there are also contours of constant R-ratio (load, stress or strain ratio). Examples of these contours in the peak vs. valley table are highlighted above. The diagonal from the upper left to lower right is the contour for R=1, peak equals valley. The row at a valley of zero is R=0. And the column under a peak of zero is R=∞. All contours radiate out from the intersection of the column for a peak of zero and the row for a valley of zero.

The R-ratio contours are useful in whether the cycles in a spectrum are within the bounds of the available data. Fatigue data is usually grouped in terms of constant R. The minimum and maximum R-ratios at which data exists start to define the boundaries of the region where damage estimations are interpolations vs. extrapolations. The largest and smallest maximum stresses (or loads or strains) used in generating the test data complete the boundaries. This concept will be demonstrated later with an example.



In the range vs. peak table, the constant R-ratio contours are more complicated. The first column is R=1. The row at a peak of zero is R=∞. All other contours radiate out from the intersection of the first column and the row at a peak of zero.

Australian-Canadian IFOST
CF/A-18 Full-Aircraft Fatigue Test

Developing Wing Test Spectrum

- Not just a classical Maneuver Spectrum
- Significant Numbers of Buffet Cycles
  - Measured load spectra for Leading & Trailing Edge Control Surfaces

Assessing Damage Contributions

- Need to reduce 500,000+ cycles for 300 hour block to a reasonable size for quasi-static testing.
  - Australians have a resonant loading system which makes it possible to apply this magnitude of cycles.
- Desire to complete testing prior to fleet retirement
- Truncation studies - Assumptions - Analysis - Coupon Test

12/1/97 APES, Inc. 8

The Canadians and Australians are cooperatively conducting full scale fatigue tests in support of their F/A-18 fleet. Based upon their fleet experience, they felt that using classical maneuver spectra would be insufficient to represent the actual aircraft environmental. Flight test data obtained from instrumented aircraft indicated a significant amount of aerodynamic buffet cycles were being induced into the wing and control surfaces. The magnitudes and number of buffet cycles were not included with the original design conditions and fatigue spectra. To maximize the benefits from these full scale tests, it is important to include the damaging buffet cycles realized by actual aircraft in the test spectra.

The raw flight recorded spectra contained over 500,000 cycles for about 300 flight hours. Such a large number of cycles can not be applied quasi-statically in any reasonable amount of time, or at a reasonable price, to a full-scale test with a goal of 18,000 to 24,000 equivalent flight hours. Truncations studies were performed to determine the non-damaging cycles. The approach for determining non-damaging cycles relies heavily on existing fatigue theories and methods. However, state-of-the-art fatigue methods do not adequately account for buffet-fatigue interaction. Thus, a small coupon test program was also conducted to confirm the analytical truncation results. APES, Inc., performed supporting studies using both strain-life and crack growth analyses. These analyses were performed using occurrence and damage tables as described in this presentation.

An Example of Spectrum Editing:

Australian-Canadian IFOST CF/A-18 Full-Aircraft Fatigue Test

- Using the Wing Fold Lug Spectrum for Illustration
  - Baseline filtered to 1.3% rise/fall level
  - Baseline filtered to 5% rise/fall level
  - Baseline filtered to 15% rise/fall level and a deadband of 30%
  - Maneuver only filtered to 15% rise/fall level, a deadband of 30%, and amplified by the corresponding dynamic loads in the baseline spectrum.

- Assessment of the Impact of Buffet on Fatigue
  - Material: 7050-T7451 aluminum
  - Crack Initiation: fraction of total Miner's Rule sum in a single spectrum pass for each cycle type defined in the table
  - Crack Growth: fraction of total crack extension in 300 hrs. starting at a 0.005 in. crack.

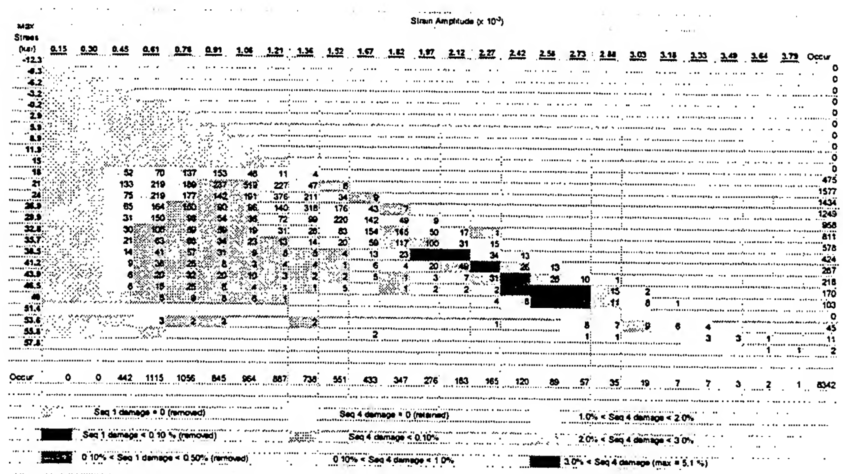
12/1/97 APES, Inc. 9

The Wing Fold Lug Spectrum is used in the following examples. Some basic truncation assumptions were selected by Canada and the impact of cycle elimination were compared to the full spectrum containing all cycles. The baseline spectrum is the flight test recorded spectrum containing both maneuver and buffet loading. The baseline spectrum was filtered based upon several different rise/fall criteria. Peak-valley sequences with rises or falls less than the prescribed value were removed from the spectrum. For the third and fourth spectra, any peaks or valleys with magnitudes less than 30% of the reference (max. maneuver load) were removed.

Both strain-life analysis, sometimes referred to as "crack initiation", and crack growth analysis were performed by APES. Specimen life predictions were made for the coupon test program along with the damage assessment presented in this paper.

On the next two pages are the results of spectrum editing for the Wing Fold Lug Spectrum. While the damage table approach was not used to edit the spectrum, we were able to determine how damaging the cycles that had been removed were.

Spectrum Editing Based on Crack Initiation



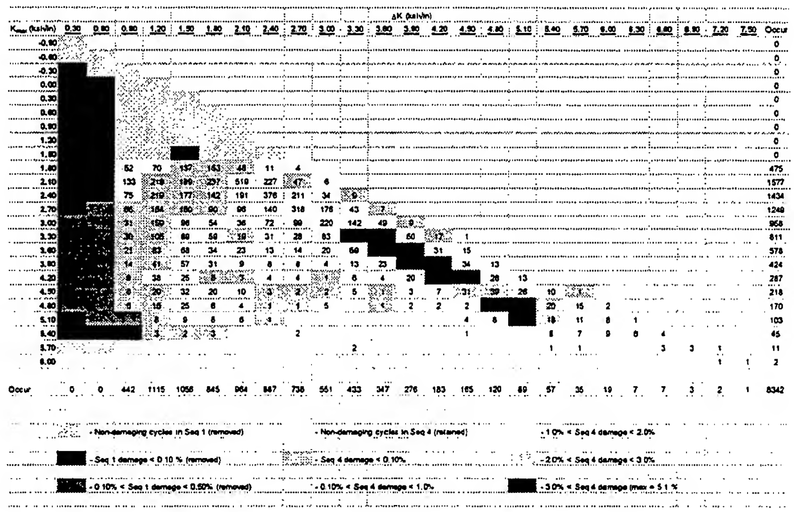
12/1/97 APES, Inc. 10

This is the occurrence table for the Seq 4 spectrum. The colored boxes without any numbers are where cycles in the baseline spectrum were removed in going to Seq 4. The cycles that were removed were not damaging in strain-life analysis. As noted earlier, the least damaging cycles are generally in the upper left of a range vs. peak table.

There are additional cycles that could be removed from Seq 4. These are the lighter boxes adjacent to where cycles removed from the baseline spectrum. Strain-life analysis indicates that these cycles are non-damaging. However, they could be damaging in crack growth.

The area of most damage are the darker boxes in the lower middle of the matrix.

Spectrum Editing Based on Crack Growth



12/1/97 APES, Inc. 11

This chart is the occurrence table for the Seq 4 spectrum at a crack length of 0.005 in. Damage is the percentage of the crack extension during a single application of the spectrum that can be attributed to cycles of a particular type.

The shaded boxes without numbers are cycles in the baseline spectrum which were removed in creating Seq 4. Some of the cycles removed were slightly damaging in crack growth, though strain-life analysis indicated no damage. Similarly, some of the Seq 4 cycles which appeared non-damaging from strain-life analysis are slightly damaging in crack growth.

Note the region of most damaging cycles is about the same in this case.

Identifying Limits of the Analysis

- Determine amount of damage from cycles outside the usual theoretical framework
  - Cycles with negative peak stresses
  - Equivalent Strain Equations
- Determine amount of damage from cycles beyond the bounds of the baseline data.
  - Fatigue analysis methods are based on a semi-empirical framework. Interpolations are more accurate than extrapolations.
  - High positive R-ratios or very negative R-ratios
  - Small amplitude cycles and Threshold region
- Determine if more baseline data is required (and what type) to provide confidence in life prediction.

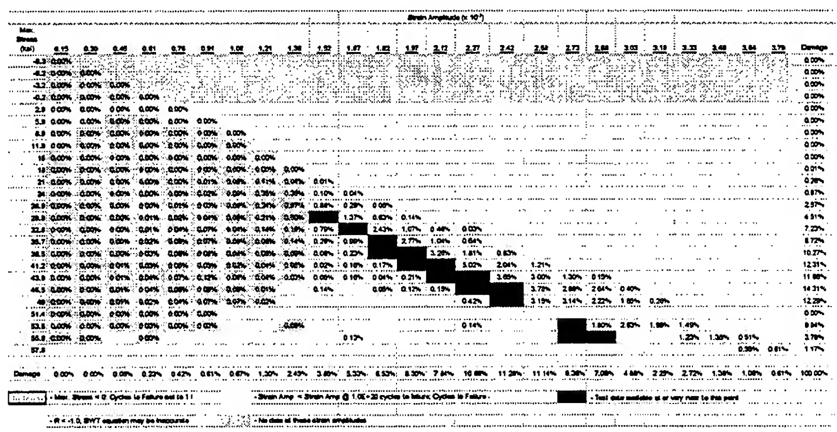
12/1/97 APES, Inc. 12

Using the same Wing Fold Spectrum, we can now identify regions where the current theories and methods breakdown. The occurrence and damage tables with the combination of peak, range, valley, and R-ratio characteristics provides a description of the type of cycles contained within the spectrum. For example, compression-compression cycles are readily identified. In addition, these tables can be used in studies to assess the impact on the damage calculations of different analysis procedures, i.e., equivalent strain equations, crack closure models, etc..

Often engineers are unaware of what material data is supporting the analysis. These tables can be used to determine when the operating conditions (loading) are outside the available baseline data. Life prediction methodology differences, strain-life vs. crack growth, can be made based upon relative damage contributions with respect to cycles to assist in advancing the total life predictions.

The engineer gains an appreciation of how accurate the life prediction is. And if important to the integrity of the aircraft, can pursue additional baseline data or improvements in the theoretical framework.

Analysis Limits for Crack Initiation

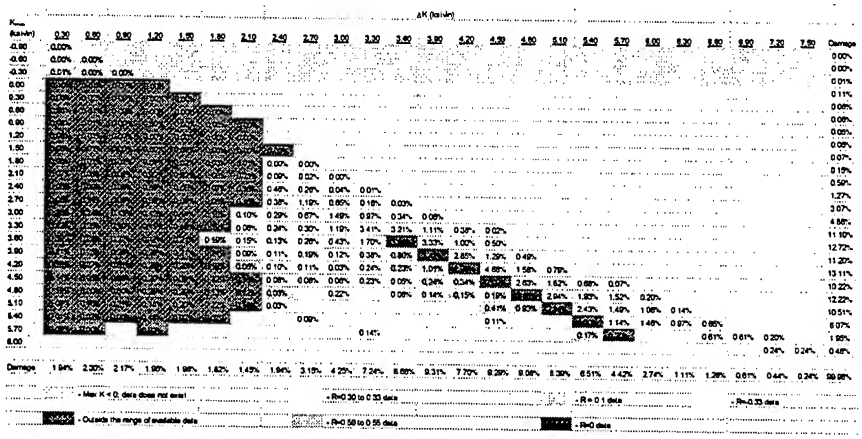


12/1/97 APES, Inc. 13

The strip of dark shaded boxes in the lower center of the chart is the region where data for 7050-T74 exist. These data are at a stress ratio of 0. There are also data at R= -1.0 and 0.5. So the right-hand portion of the chart is all interpolation.

The shaded boxes at the lower left represent a region for which no data exists. These are primarily small strain amplitude tests which would have very long lives. Special test techniques (high frequency test machines) are needed to obtain data in this region.

Analysis Limits for Crack Growth



12/1/97 APES, Inc. 14

For crack growth also we see that the lower right portion of the chart is in the region where data are available. (This is at a crack length of 0.005 in.) Thus, the majority of damage is calculated by interpolating between available data.

There is a large region at the left side of the graph (small amplitudes) where data are not available. If there were significant fatigue drivers in this region, it would be necessary to obtain data in this region in order to have confidence in the life prediction.

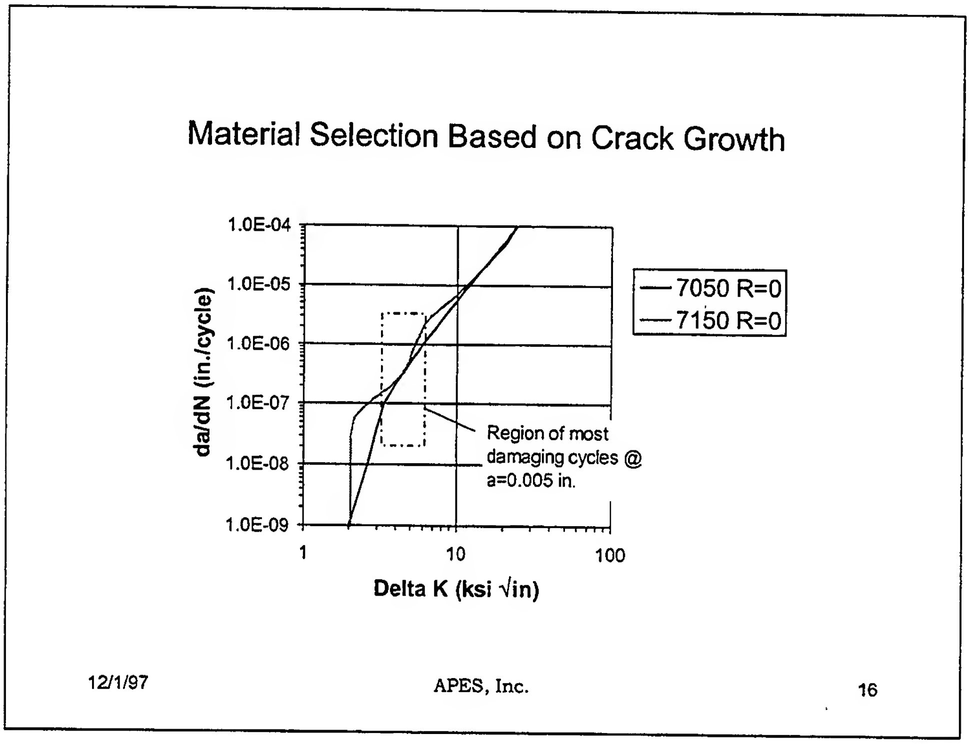
Proper Material Selections

- Design Requirements for Aircraft Require New Approaches to meet proper and optimum Material Selection
  - Requirements for all air vehicles (military, commercial, and general aviation) have durability & damage tolerance requirements
  - Lighter, Cheaper, and Longer Lasting
- For materials with similar strengths, analyst can identify what types of cycles are fatigue drivers.
  - Choose between materials based upon their response/properties under the critical conditions.
  - Identifying the limits of existing data compared to the drivers can establish confidence in selection.

12/1/97 APES, Inc. 15

Durability and Damage Tolerance requirements for the design and maintenance of aircraft have been extended into all types of air vehicles. Fatigue and fracture criteria control the sizing and geometric details of many components. Proper material selection thus requires a means of comparing material for the optimum selection to meet the requirements. The use of the damage tables assists in making the appropriate decision through a better understanding of the spectrum contents and the damage accumulation attributes.

One of the ways the damage tables can be used for this type of evaluation is illustrated in the following chart.



For crack growth from a crack of 0.005 in., the major damaging cycles in the Wing Fold Spectrum are around R=0 between ΔK of 3.0 ksi√in and 6.0 ksi√in. If 7150-T7751 plate were being considered as a replacement for 7050-T7451 plate in this location, and we were concerned about crack growth at small sizes, it is obvious that 7150 would not be a good choice. In this region, 7150-T7751 generally has faster crack growth rates than does 7050-T7451.

Conclusion

- Fatigue analyses sometimes are more than just a life prediction.
  - An understanding of the spectrum and how it interacts with the material properties are frequently needed for sophisticated solutions to fatigue problems.
- Damage Tables and Occurrence Tables are tools that can be used to engineer solutions to fatigue problems.
  - Identify what cycles are primarily responsible for fatigue damage.
  - Enable selected editing of fatigue spectra for testing.
  - Select equivalent damage cycles to replace high cycle counts
  - Establish when the loading is outside the envelope of known fatigue problems.
  - Determine if an alternate material will solve the problem.

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Significant benefits for the entire aircraft life cycle can be achieved by utilizing as much information and acquiring insights in all aspects of the fatigue process. The tables and tools presented here provide further understanding of the characteristics, contents, and damage contributions in a fatigue spectrum. These are a few of the techniques that will assist in improving fatigue analysis. Truncation and equivalent damage cycle replacement procedures will assist in more successful and meaningful full-scale fatigue tests. Significant cost benefits will be realized in combination with more confident decisions. Improvements in the design and /or repair of structure can also be enhanced through the simple procedures and processes presented in this paper.

Effect of Compression on 7075 Aluminum Alloys

Presented at ASIP

3 Dec 1997

By

James A. Harter

AFRL/VASE

and

John H. Elsner

Analytical Services & Materials

Objective

- Determine the effect of compression on crack growth rates in 7075-T6 and T73
- Observe the effect of various maximum stress levels at low R values
- Observe the effect of humidity on these data

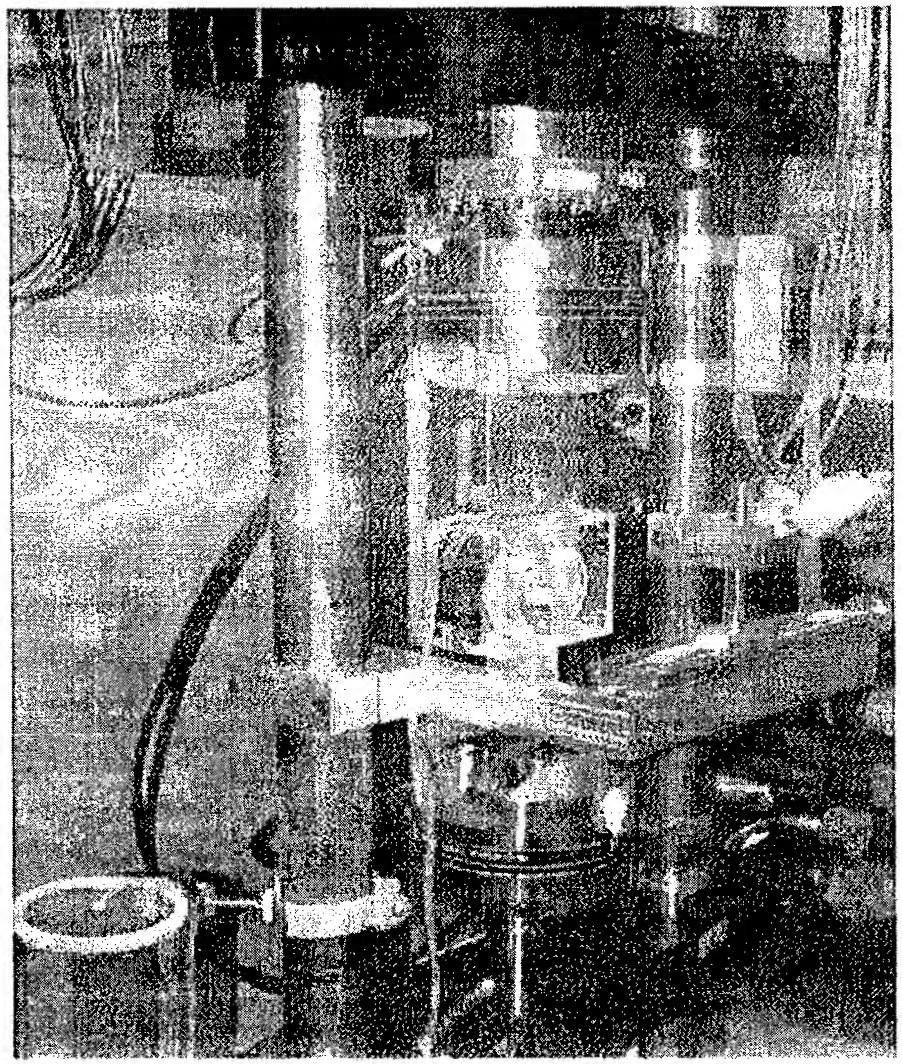
Approach

- MT test specimens ($W = 3.9$ in., $L = 16$ in., $T = 0.25$ in.)
- Perform constant stress level crack growth rate tests at $R = -0.5, -1.0, -6.0$, and -9.0
- Maximum remote load levels: 5.0, 22.5, 30.0, and 38.6 kips
- Cyclic frequency (6-10 Hz)
- Controlled Humidity (<15% R.H. - 7075-T7351)
- Lab Air Tests (7075-T651)

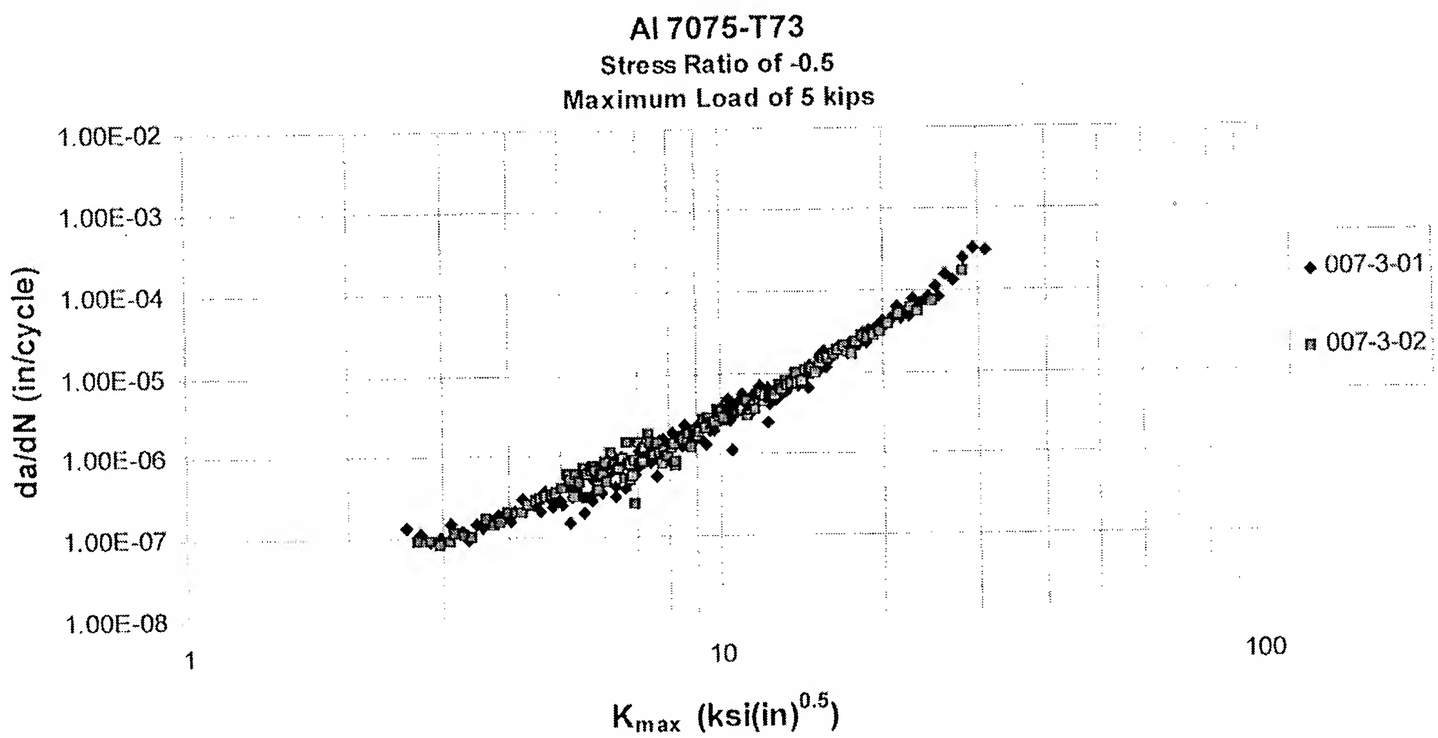
7075-T7351
Controlled Humidity Data

< 15% R.H.

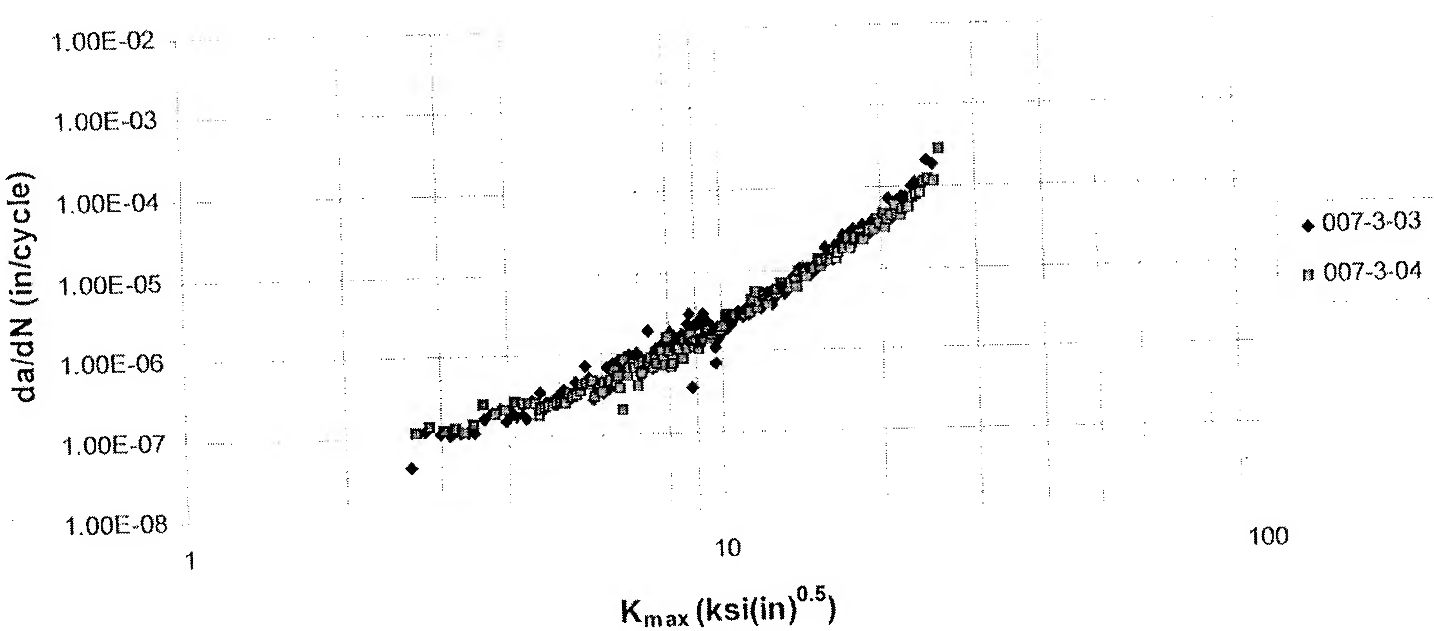
Humidity Control



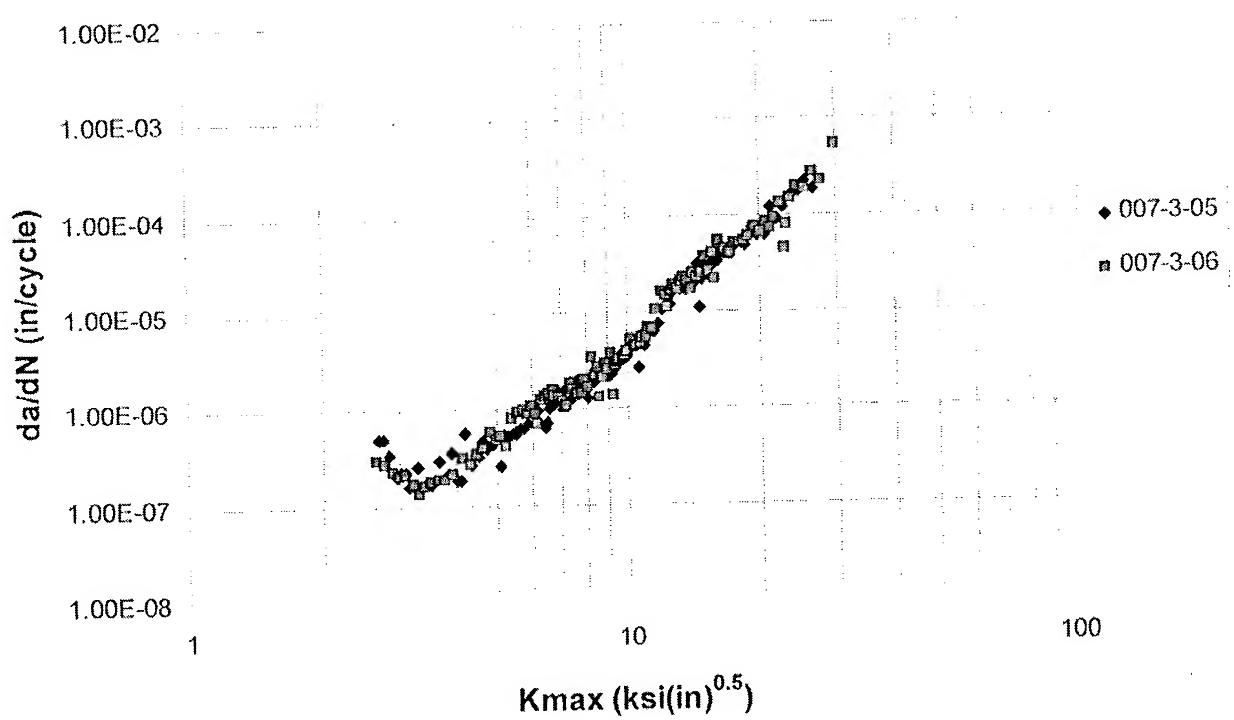
Pmax = 5 kips, R=-0.5



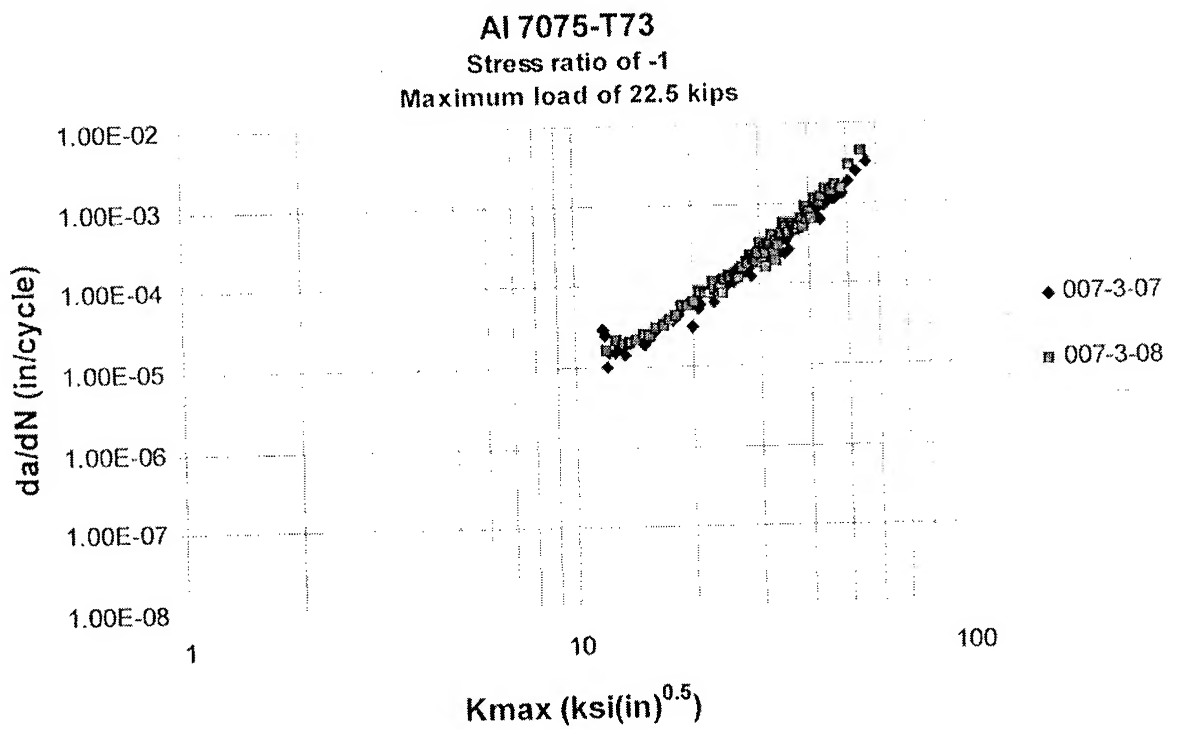
Pmax = 5 Kips, R=-1.0



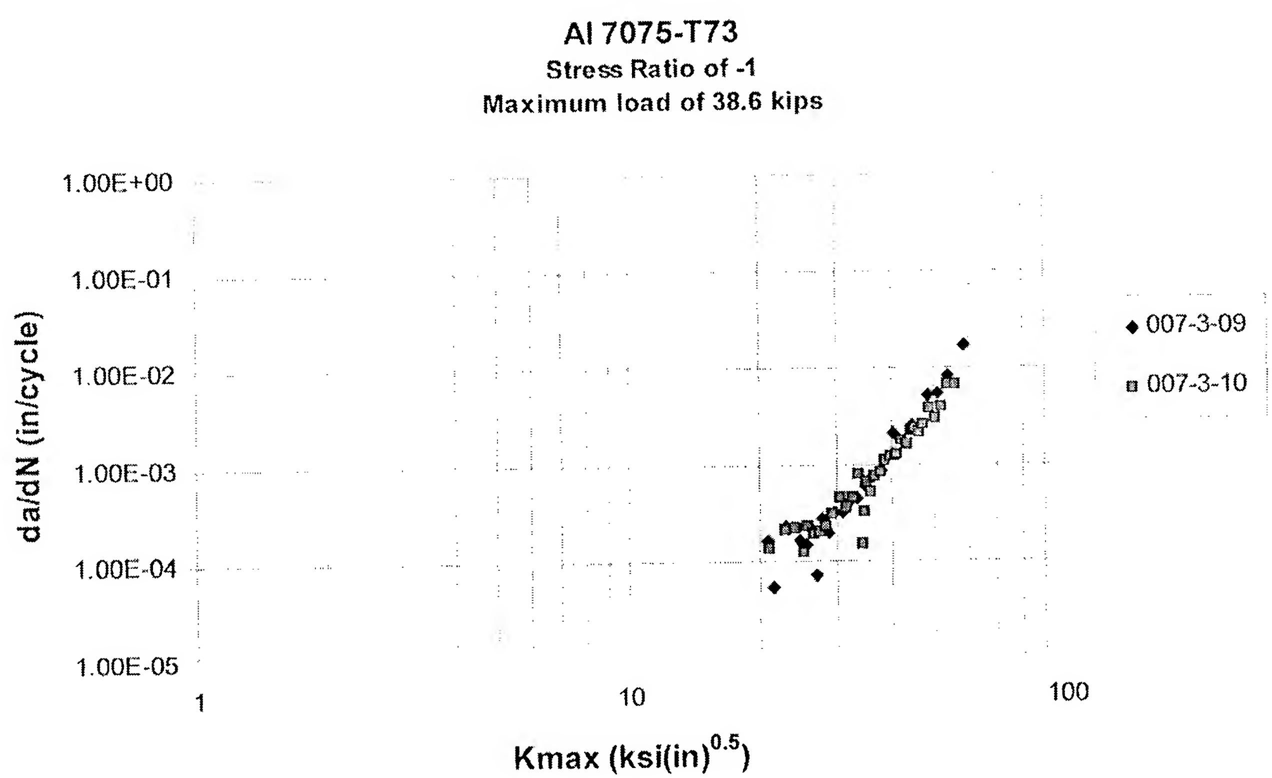
Pmax = 5 Kips, R=-6.0



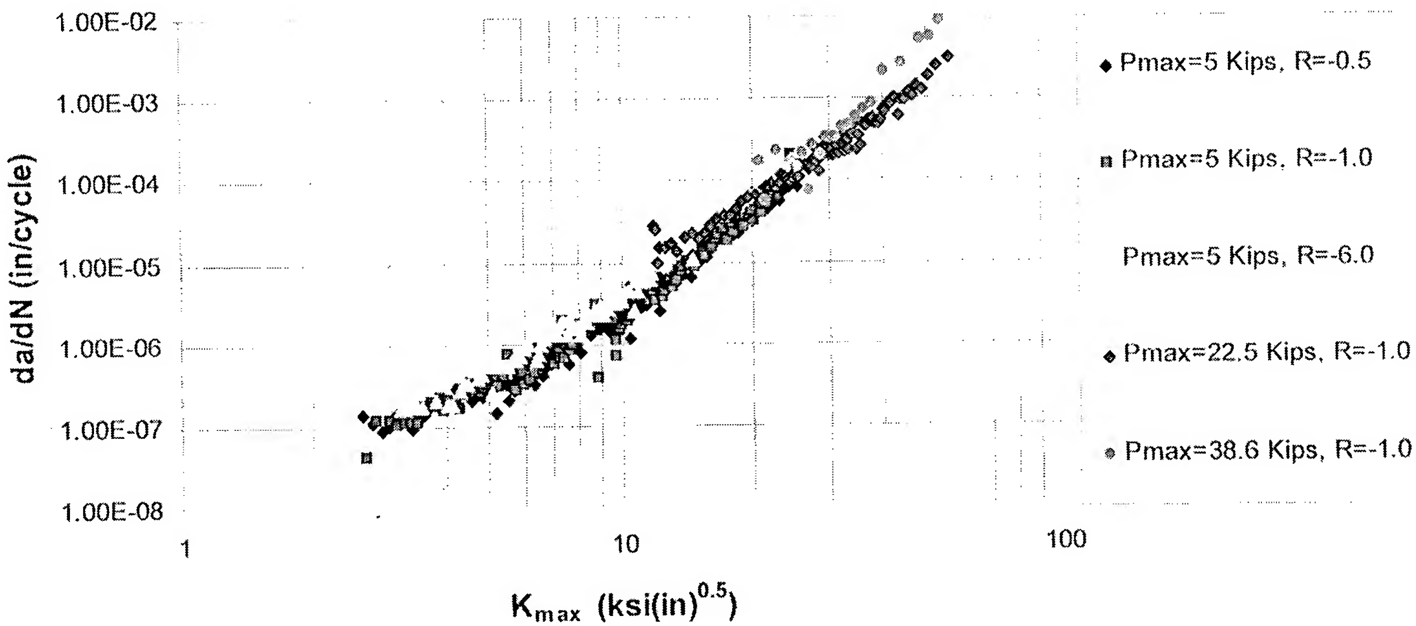
Pmax = 22.5 Kips, R=-1.0



Pmax = 38.6 Kips, R=-1.0

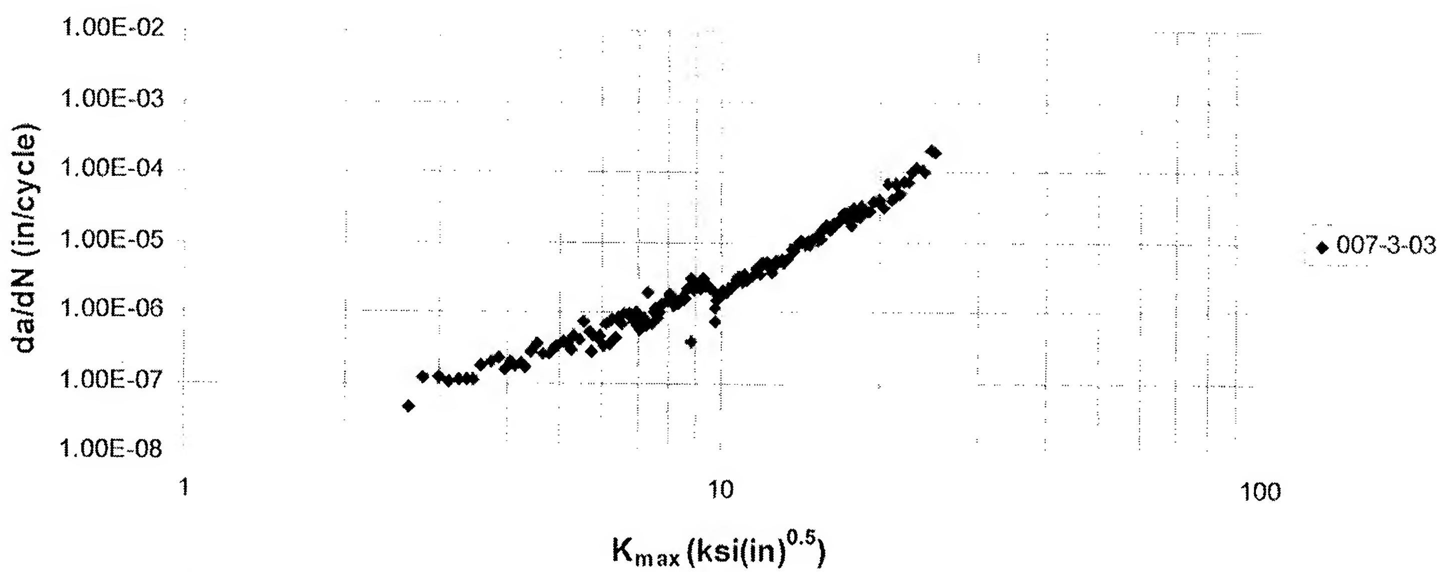


R.H. < 15%



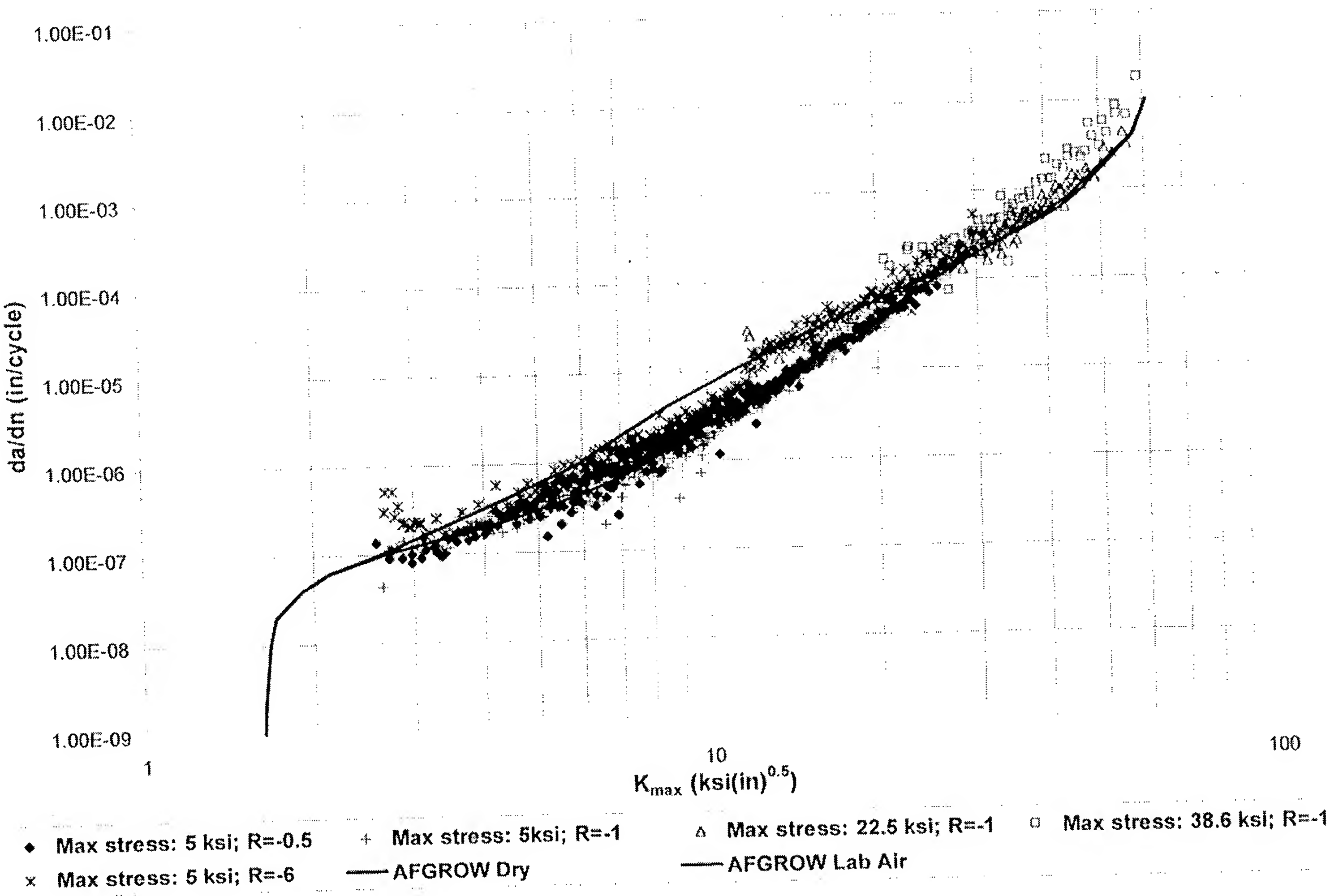
1.00E-02
1.00E-03
1.00E-04
1.00E-05
1.00E-06
1.00E-07
1.00E-08
da/dN (in/cycle)
1
10
100
K_{max} (ksi(in)$^{0.5}$)
Pmax=5 Kips, R=-0.5
Pmax=5 Kips, R=-1.0
Pmax=5 Kips, R=-6.0
Pmax=22.5 Kips, R=-1.0
Pmax=38.6 Kips, R=-1.0

Data Variation for R.H. < 15%



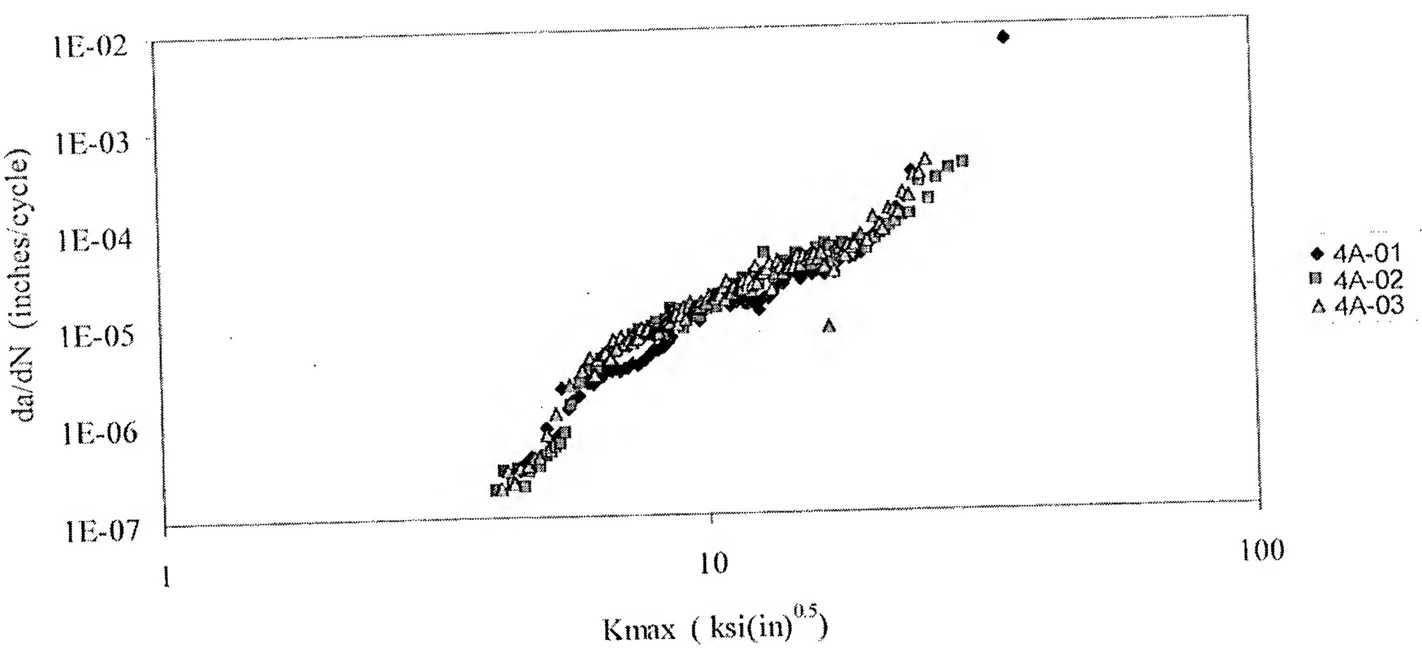
Desiccant Changed During Test When R.H. Reached 15%

7075-T73
R.H. < 15%   Rlow=-0.33

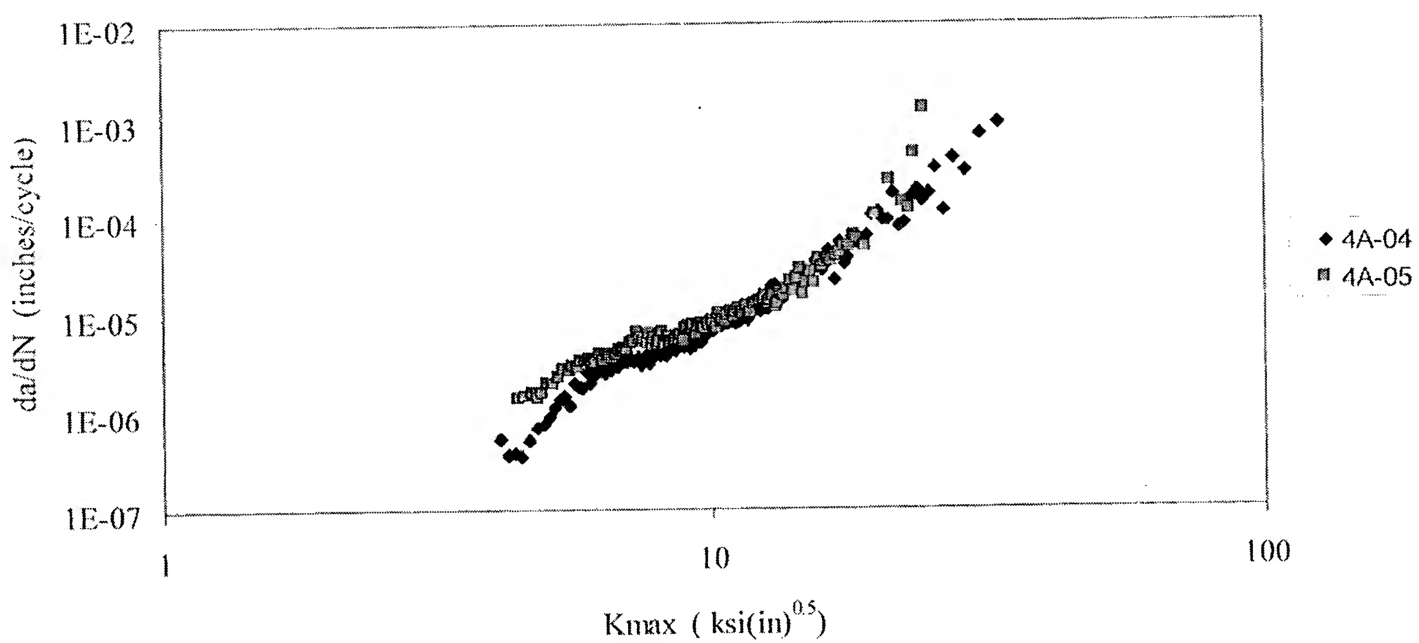


7075-T651
Lab Air Tests

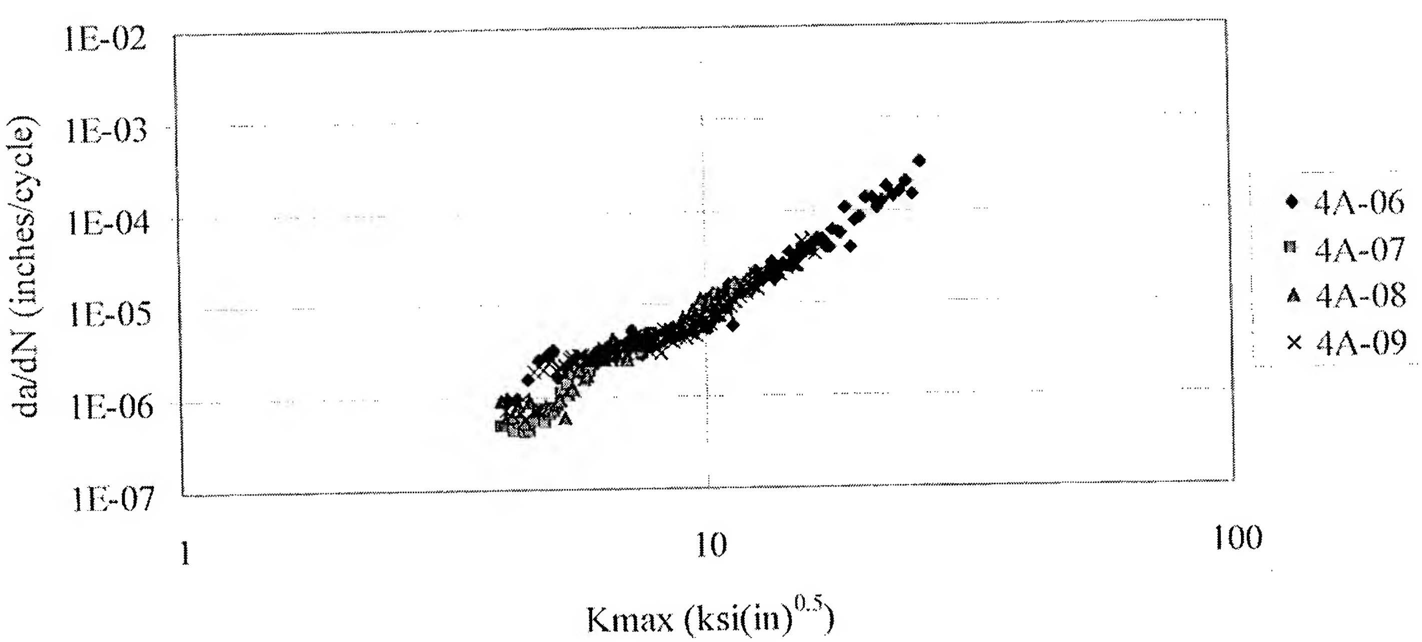
Pmax = 5 Kips, R=-0.5



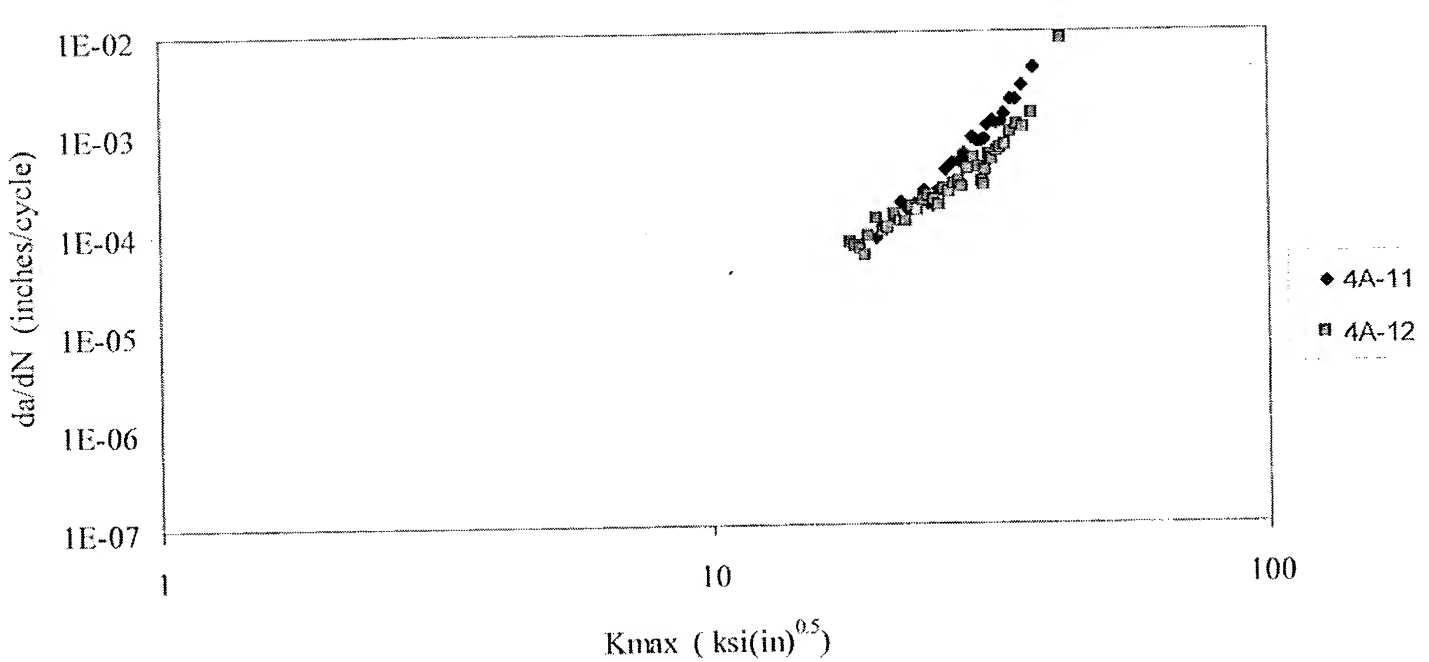
Pmax = 5 Kips, R=-6.0



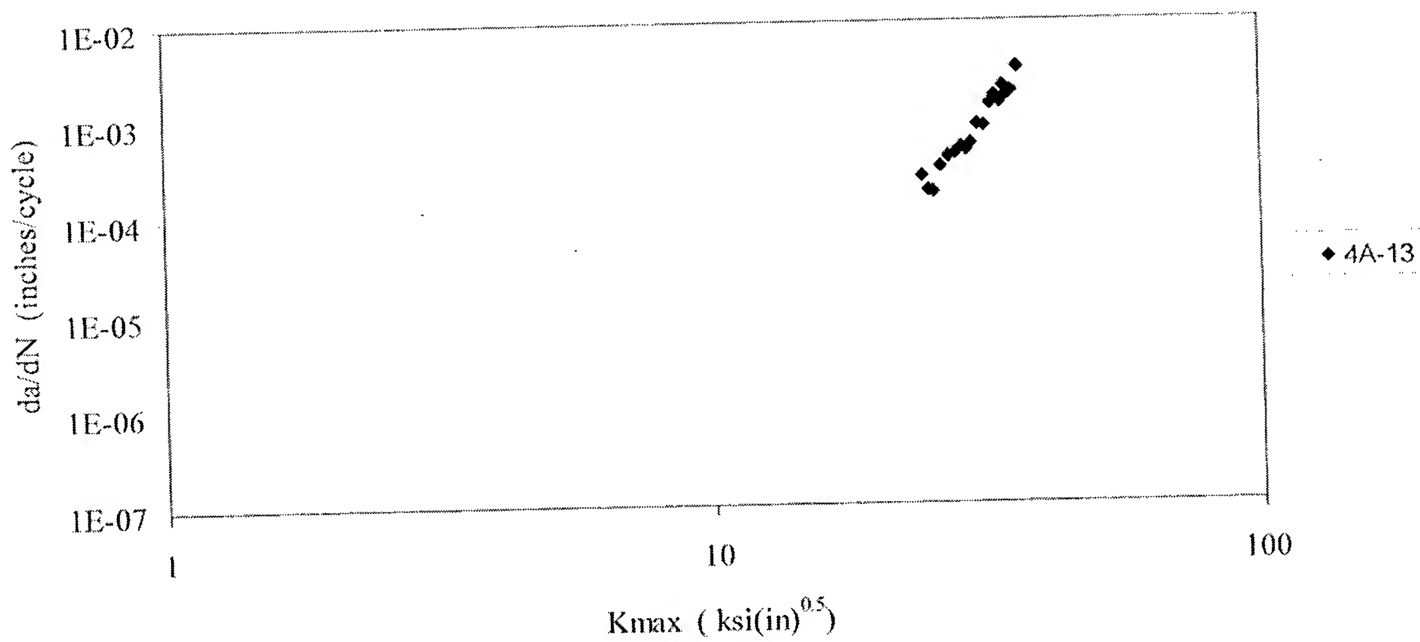
Pmax = 5.0 Kips, R=-9.0



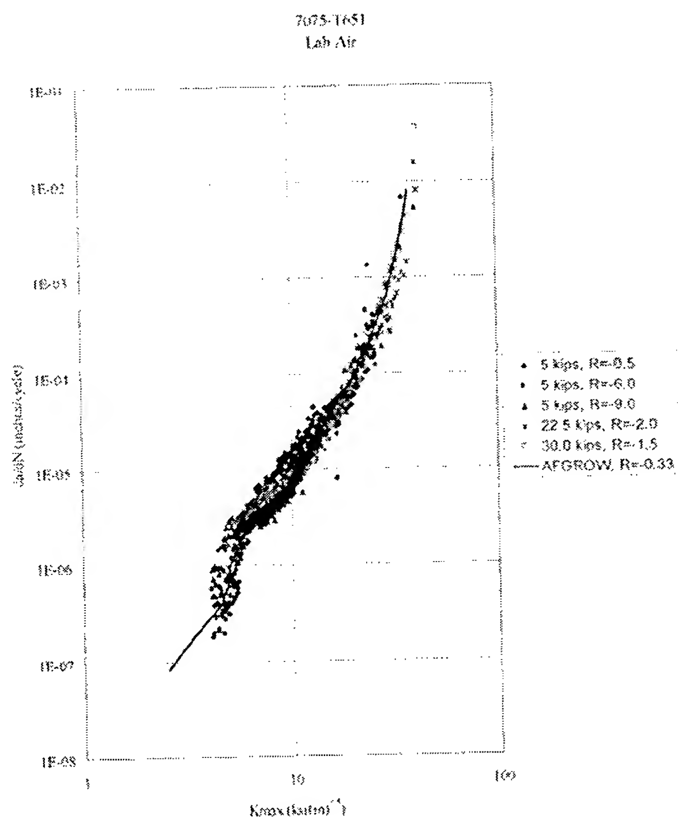
Pmax = 22.5 Kips, R=-2.0



Pmax = 30 Kips, R=-1.5



7075-T651, Lab Air Data
Rlow=-0.25



Comments

- Constant loading was used since it was felt to be more representative of real crack growth problems (without introducing the complexity of random loading)
- These data compared well with previous data obtained in-house for these materials
- Data scatter makes it difficult to make decisive conclusions; **however**, trends are indicated
- Compressive stresses clearly affect local stress levels near the crack tip
- Funding was not available to do more testing
- Constant Kmax tests would have provided valuable data when viewed in conjunction with these data
- Constant Kmax tests would have shown any differences in rate for different unloading ratios

Conclusions

- Da/dN vs. Kmax does not appear to vary significantly for R < Rlow for 7075-T7351 or 7075-T651
- While the data for T7351 shows some acceleration effect at a Pmax of 38.6 Kips, this is not indicated for T651 at 30 Kips.
- Humidity effects are shown to play a role in the crack growth rate data scatter for these alloys (even below 15% R.H.)
- More tests should be conducted either in a vacuum or under lower humidity levels and/or very high cyclic frequencies
- Constant Kmax test data may provide additional valuable insight to this issue

LUNCH PRESENTATION

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A Risk Based Approach to Manage Critical Locations on a CF/RAAF F/A-18 Fatigue Test Article

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Introduction

Under the auspices of the F/A-18 International Follow-On Structural Test Programme (IFOSTP), the Canadian Forces (CF) and the Royal Australian Air Force (RAAF) are jointly conducting a series of full scale fatigue tests on the F/A-18A/B aircraft. These tests will help determine the safe and economic life of the airframe, certify Engineering Change Proposals (ECP's) and generate engineering data to assist in the management of the two fleets. Several factors contributed to raise the requirement to undertake an expensive testing programme instead of relying on the certification of the basic airframe and ECP's provided by McDonnell-Douglas (now Boeing), herein referred as the Original Equipment Manufacturer (OEM).These factors are:

a. Design Pedigree. The F/A-18 was designed under a safe life philosophy where failure, considered as the initiation of a 0.010 inch flaw, should not occur during the 6,000 hour service life of the aircraft. A scatter factor of 2.0 on durability was used for all component and full scale fatigue tests. The use of the safe life philosophy in the design and certification of the aircraft has resulted in several virtually non inspectable, highly loaded components showing fast crack growth properties. By contrast, the CF and the RAAF have generally adopted a Damage Tolerance approach to certify their aircraft. Although both countries realise that fracture critical locations on the F/A-18 cannot be managed using Damage Tolerance, there may be room for using the approach at non critical locations.

b. Scatter Factor. The OEM certified the 6,000 hour safe life of the aircraft using a scatter factor of 2.0, demonstrated by the successful completion of 12,000 Simulated Flight Hours (SFH) of testing under a severe 95 percentile usage spectrum. The CF and the RAAF, on the other hand, require that a scatter factor of 3.0 be used in

durability tests of aircraft fitted with individual fatigue tracking systems. As a result, certification of the airframe for the CF and the RAAF requires the completion of 18,000 SFH of fatigue testing.

c. Usage Severity. In the mid to late 1980's, the CF fleet started to experience structural problems which, according to the certification tests, should not have occurred until much later in the service life of the aircraft. An initial review showed that the CF fleet consumed fatigue life at a faster rate than that simulated by the OEM fatigue tests. A review performed for the newer RAAF fleet suggested that the Australian usage rates were closer to the CF's than to the USN's design, indicating that their fleet would also experience premature structural problems.

d. Engineering Changes Proposal Certification. The various fatigue tests performed by the OEM exposed several major structural problems which have been addressed through in-service retrofits (ECP's). Many of these ECP's addressing fracture critical components, were never certified on full scale fatigue tests. The certification of these ECP's generally rests on analyses supported by coupon testing for a design life of 12,000 SFH. The severe CF and RAAF usage combined with a higher scatter factor requirements raised concerns that these ECP's may not last the intended 6,000 hour service life.

Through agreement between Australia and Canada, three separate full scale fatigue tests have been mandated to address the concerns raised previously. To ease the interpretation of the test results into fleet management actions, the three tests apply actual usage spectrum representing one year of flying, instead of applying a design spectrum. The IFOSTP tests are defined as follows:

a. FT-55 is a quasi-static fatigue test of the centre fuselage structure (figure 1). It is conducted at the Bombardier Defence Service Division (BDSD) site in Mirabel, Québec;

b. FT-46 tests the F/A-18 aft fuselage and empennage. It is a unique full scale fatigue test in that pneumatic shakers superimpose dynamic loads in real time to the quasi-static manoeuvre loads applied by hydraulic actuators. The Aeronautical and Maritime Research Laboratories (AMRL) in Melbourne, Australia, has recently completed the pre-LEX fence usage phase of the test and is making preparation for the post-LEX phase; and

c. FT-245 will address the F/A-18 wing critical components. It will be a quasi static tests, with allowances made to account for the effects of dynamic loads on some components. The Institute of Aerospace Research (IAR) will conduct the test in Ottawa, Canada.

FT-55 Status

FT-55 has accumulated 12,000 SFH of representative usage as of mid-August, 1997 and is now undergoing an extensive inspection. This is a significant milestone for the CF and the RAAF in two ways. First, the existing 4,000 hour certification for the Canadian fleet has been demonstrated. Second, due to the severity of its spectrum, one can argue that FT-55 has now accumulated as much wing root dominated damage as what was demonstrated by the ST-16 full scale test. ST-16, the OEM test which certified the centre fuselage section for the U.S. Navy, accumulated approximately 16,000 SFH of wing root dominated damage. Many of the ECP's developed as a result of ST-16, and incorporated on FT-55 have also been demonstrated to a rough ST-16 equivalency.

Through the first 12,000 SFH of testing, regular inspections of the test article have allowed the discovery and airworthy disposition of defects previously reported during ST-16. As anticipated, defects have shown a tendency to appear on FT-55 at an earlier time than observed in previous fatigue tests. The 12,000 SFH inspection is now complete and the test article undergoing repairs and modifications. The break down of the maintenance actions is as follows:

a. 12 major repairs;
b. 21 minor repairs; and
c. 7 preventive modifications

The test is currently scheduled to re-start in May 1998, and reaching the 18,000 SFH is expected to take an additional 1.5 to 2 years.

The Dilemma

FT-55 has been operating so far with the benefits of knowing where the ST-16 failures had occurred and, based on crack initiation time comparisons using the two spectra, roughly when to start looking for them. In the next test phase (12,000 to 18,000 SFH), "ST-16 tear down type" failures will continue to develop and, if left un-addressed, may develop into significant problems for FT-55. Additionally, some of the OEM ECP's designed for a 12,000 SFH life will likely initiate failures of their own.

Under directions from the CF, BDSD performed crack initiation analyses on known failure sites which have high safety of flight implications. Ten locations, identified in table 1 and illustrated in figures 2 through 4, were determined to have the potential to create problems on FT-55 before the 18,000 SFH milestone. This analysis was limited to the wing carry through bulkheads because they are generally highly stressed under normal usage and also because they contain features with very small critical flaw sizes and subject to rapid crack growth once a defect has initiated. Many critical locations are non-inspectable on a fleet aircraft and, in some cases, even on the test article.

The BDSD spectrum comparisons and crack initiation analyses anticipated that some critical locations should have initiated 0.010 inch flaws by the 12,000 SFH milestone, and that others will do so before reaching 18,000 SFH. Given the required scatter factor of 3.0, there is therefore a significant risk that certification of these locations will not be achieved on FT-55. In addition to not being able to certify a given critical location, there is also an added risk to FT-55 that the fracture of a wing carry through bulkhead would grossly yield, or fracture, the two remaining carry through bulkheads and distort further test results in the centre barrel.

A solution envisaged for this problem consists of designing preventive modifications to address all perceived critical areas for incorporation during the 12,000 SFH inspection. This course of action would result in discarding test failure information and commit the CF and RAAF fleets to incorporate these modifications without having proven that they are really needed. Although the stated goal of a 6,000 FH certification would then be more likely to be achieved, it would not be done in a cost effective manner. This solution is obviously not acceptable for the IFOSTP partners.

Another possible solution consists of incorporating preventive modifications to one side of the aircraft and leaving the other side develop the flaw that proves the need for the modification. This would preserve test point information and evaluate prototype retrofits. This solution, however, does not account for the risk of an undetected flaw rupturing a wing carry through bulkhead and possibly cause the premature end of the test.

The dilemma between the conservation of test points, the need to demonstrate 18,000 SFH on FT-55 and the test article safety results in the requirement for a risk based methodology that can assess modification candidates and recommend a course of action in a systematic and repeatable fashion. The next section explains the risk methodology used on FT-55 to assign a risk factor to each critical location, make initial recommendation and assess risk mitigation factors to determine the final course of action. Following the risk methodology, three examples will show how the methodology was applied to FT-55. Table 2 provides a summary of the risk assessment for all ten locations shown in figures 2 to 4, and table 3 presents a summary of the planned course of action for FT-55.

Risk Methodology

General. Reference 1 defines risk as "the probability of occurrence of a negative outcome multiplied by the consequence of that negative outcome". The method developed for FT-55 has been adapted from procedures used at the CF Aerospace Engineering Test Establishment to evaluate safety risks for flight test programmes. The FT-55 process is illustrated on figure 5. The process evaluates risk using two different sets of parameters: one set addresses fleet management issues and the other test article safety. The two streams join for the assignment of an overall risk factor and a initial recommendation for modification. The following paragraphs expand on the parameters used to identify and quantify the risk.

Risk Identification. The identification of the risk on FT-55 comes from the analysis of ST-16 test results, in terms of failure times, the consequence of these failures for that test and the suitability of the modification (ECP) proposed by the OEM to reach the 18,000 SFH milestone under the FT-55 load spectrum. Defects identified during the tear down inspection and for which no ECP's have been proposed also constitute significant risks for FT-55.

FT-55 Probability of Failure (POF): From a test article management perspective, the probability that a flaw could grow to critical size before being detected constitutes the negative event that must be controlled. The assignment of a probability of failure is determine by an evaluation of accessibility for inspection on FT-55, the geometry of the feature, the critical flaw size and crack growth information if available. For example, an area subject to fast crack growth rate and not accessible for inspection, such as the wing attachment lugs, would be assigned a high probability of failure.

FT-55 Consequence of Failure: ST-16 test information is used to evaluate whether a failure would compromise test results locally (i.e. no effect on other potential test points), globally on one or both sides of the aircraft, or area wide such as a in wheel well or on a longeron. A global impact results in the assignment of a high consequence for the test article, an area wide impact in the assignment of a medium consequence and a local impact in the assignment of a low consequence.

FT-55 Risk Factor. The probability and consequence of failure are combined in figure 6a to establish a risk factor that is either low, medium or high for the test article.

Fleet POF: Failure in a fleet management scenario consists of the probability that some aircraft will initiate a 0.010 inch flaw during their 6,000 hour service life. Crack initiation life predictions and the required scatter factor set the probability of failure from the fleet management point of view. In that context, features showing predicted crack initiation lives of less than 12,000 SFH are assigned a high probability failure, while 18,000 SFH predicted crack initiation lives result in the assignment of low probabilities of failure. Predicted crack initiation lives between 12,000 and 18,000 SFH result in medium probabilities of failures being assigned.

Fleet Consequence of Failure: The consequence of failure considers the impact of a 0.010 inch flaw on given structural feature. The low, medium or high factor assignment depends in this case on the accessibility of the area for inspection on a fleet aircraft, the size of the critical flaw and crack growth rates. The initiation of a flaw in a difficult to inspect and highly loaded feature would likely result in the loss of the aircraft, and result the assignment of a high impact factor. In a structure that is fracture critical but inspectable, the chances that the flaw would be picked up before it becomes critical are better and a medium impact is therefore assigned. Finally, a flaw initiating in an inspectable area and that is suspected of exhibiting slow crack growth or self arrest properties will be assigned a low impact. It is important to note here that the fleet consequence of failure assessment is used to reach a decision to modify the FT-55 test article. It does not reflect on how the locations will be managed in service.

Fleet Risk Assessment. Figure 6b shows how the probability and consequence of failures are combined to assign an overall fleet management risk factor.

Initial Recommendation: Figure 6c finally combines the test and fleet management risk assessments to make an initial modification recommendation for the test article. The reader will notice that this matrix is biased towards reducing the risk to the test article and accepting more risk from the fleet management perspective. This bias is acceptable since it is information coming out of the test article that will ultimately form the basis for managing the fleet.

Risk Mitigating Action. For three locations on the test articles, the probabilities and consequences of failure were high enough to warrant an initial recommendation to modify both sides of the test article. In an effort to preserve the maximum number of test points, a second look was given to these locations with the purpose of reducing the risk by improving access for inspection, or considering alternate means of gaining certification.

Plan of Action. The final step of the risk analysis consist of evaluating initial modification recommendations and the suitability of proposed risk mitigation actions. A final recommendation regarding the incorporation of preventive modifications is then finally made.

Case Study # 1 - Bulkhead Outer Mould Line Flanges

Risk Identification. Figures 7 shows the three F/A-18 wing carry through bulkheads of a retired USN aircraft. They have been exposed to perform trials of the modification proposed for FT-55. The aft bulkhead (Y488) has suffered catastrophic failures in the area located below the lower lug during the ST-16 full scale fatigue test; in one case causing the failure of the centre bulkhead (Y470) before the test was shut down. Although the Y453 and Y470 did not experience catastrophic fatigue related failures, their geometry and loading magnitude are close enough to the Y488 to warrant concern. To address the problem, the OEM implemented a series of in-service retrofits and production line changes for the newer aircraft. Finally, the areas of concern can be reliably inspected only once the wings are removed. The CF and RAAF situation for the bulkheads is as follows:

a. The Y488 bulkheads have been shot peened in production. An in-service retrofit (ECP 45) re-profiled the outboard flange and re-applied shot peening to the bulkhead. The two shot peening operations go some way in reducing the probability of failure of the bulkhead prior to 12,000 SFH of testing, but there is uncertainty regarding the material quality and the quality of the shot peening process employed. Finally, the Y488 configuration has never been certified through full scale test; and

b. The Y470 and Y453 bulkhead have been shot peened in production. Corner cracks were detected at the ST-16 tear down inspection on the forward edges of the flanges, outside of the zones covered by the shot peening. The OEM never addressed these flaws through in-service retrofits. In light of the requirement to test to 18,000 SFH

and the severe FT-55 spectrum, these bulkheads may actually be more critical than the Y488.

FT-55 POF. Inspection of the lower lug areas on FT-55 requires the removal of the wings. The susceptibility to fast crack growth rates and small critical flaw result in a high probability that a flaw initiating in the Y488 bulkhead would reach critical size undetected. Loading and geometry similarities between the three bulkheads indicate that the Y470 and Y453 would also be subject to rapid crack growth if a flaw initiated on the OML flanges. The probability of a flaw growing to critical size undetected on either of the three bulkhead is therefore assessed as high.

FT-55 Consequence. Failure of either wing carry through bulkhead due to cracking in the OML flanges would result in a large over load being applied on the affected side and possibly result in the cascade failures of the three bulkheads. In the worst case scenario, the opposite side of the test article would sustain a substantial overload that would compromise further test results for the centre barrel section. The consequence of failure is therefore assessed as high .

FT-55 Risk Factor. Referring to figure 6a, for a high probability of failure and high consequence, the risk to the test article safety caused by the failure of the OML flanges is determined to be high.

Fleet POF: There is some variation regarding the shot peening and modification status of the three wing carry through bulkheads. Uncertainties also exist regarding the effectiveness of the shot peening applied to the bulkhead and its life improvement. Calculations using conservative assumptions indicate that crack initiation life ranges from 12,000 to 18,000 SFH for all three bulkheads. Using a scatter factor of 3.0, the probability that some fleet aircraft will develop a 0.010 inch flaw in either of the three wing carry through bulkheads within the required 6,000 FH service life is assessed as medium.

Fleet Consequence: The outboard flanges are all fracture critical features, subject to fast crack growth rates and non inspectable. Initiation of a 0.010 inch flaw in this location would result in the loss of the aircraft in a relatively short term. The consequence from a fleet management perspective is assessed as high.

Fleet Risk Factor. Referring to figure 6b, for a medium probability of failure and high consequence, the risk to fleet certification is determined to be high.

Initial Recommendation: Referring to figure 6c, an initial recommendation is made to incorporate preventive modifications on both sides of the test article.

Risk Mitigating Factors. The initial recommendation to modify both sides of the test article implies expensive retrofits for the CF and RAAF fleets at a later date. A follow-on review of the of the risk assessment considered the bare bulkhead test currently performed by AMRL

and means of improving access for inspection on FT-55. These factors can be summarised as follows:

a. The FT-488/2 bare bulkhead test will provide an additional source of data to increase the confidence in the life improvement provided by ECP 045 (CF Mod CD-022). However, a requirement for comparative coupon tests has been identified to quantify life differences between old 7050-T7451 stock (bulkhead material on fleet aircraft) and the newer stock used in the fabrication of the FT-488/2 test article.

b. The high probability that a flaw will grow to a critical size undetected can be reduced significantly by cutting a nominally non stressed portion of the drag longeron (figure 8) to allow inspection of the Y488 OML flange. Inspection windows approximately 1.0 inch in diameter can also be the duct skin and/or the outer skin to allow inspection of the Y453 and Y470 bulkheads OML flanges forward and aft edges. Finally, acoustic emissions monitoring of all three bulkheads OML flanges will continue and will hopefully provide advanced warning of a failure.

Plan of Action. Canada and Australia accepted the risk mitigating factors discussed above. Strain surveys performed on the test article, analyses and prototype trials on the retired USN fuselage have shown that reliable inspections of the critical areas are now possible and that the impact of the proposed inspection windows on surrounding structures is negligible. The plan of action for FT-55 now consists of shot peening all three bulkhead OML flanges on the RHS, and incorporate inspection windows on the LHS.

Case Study #2 - Y470 Bulkhead At X-19

Risk Identification. The Y470 - X19 (figure 9) is located in the wheel well of the aircraft. At 12,000 SFH of testing on the ST-16 test, a large crack going up the web and crossing into the adjacent bay almost severed the section. However, even with a large crack, the test article could still sustain the application of the limit load case of the spectrum. Further NDI inspections revealed several other initiation sites near the X-19 as well as in the upper outboard corner of the same pocket (X-24). The critical flaw size for the web crack was estimated through fractographic analysis as 0.400 inches. The OEM proposed a retrofit (ECP 365), which consists of shot peening the pocket and installing an aluminium doubler under the lower flange. This retrofit was supported by FE analysis and simple coupon tests to evaluate the life improvement factor provided by the shot peening. However, it was not certified through a full scale or component test. The BDSD review of the ECP 365 and further analysis raised concerns that the X-19 location would likely initiate cracking before the 12,000 SFH milestone on FT-55 unless shot peening was re-applied.

FT-55 POF: The X-19 area is located in the wheel well and is accessible for inspection. The inspection is made complex by the local geometry, the presence of previous shot peening and the relatively small size of the critical defect. However, given the frequency of the inspections and the use of LPI as a back up technique to Eddy Current, the probability that a flaw would grow to critical size undetected on FT-55 is assessed as low .

FT-55 Consequence of Failure: Failure of the X-19 on FT-55 would affect other test results in the wheel well area of the affected side. The impact is therefore assessed as medium .

FT-55 Risk Factor. Referring to figure 6a, for a low probability of failure and a medium consequence results in the assignment of a medium risk factor for the test article.

Fleet POF: Crack initiation calculation indicate that the baseline (not shot peened) X-19 would have a life of less than 6,000 SFH under FT-55 spectrum loading. Repetitive shot peening on FT-55 explains why no cracking has initiated in the area yet. While awaiting the results of an exhaustive study on the benefits of repetitive shot peening for the X-19, the risk that some aircraft will develop a 0.010 inch flaw in the area is assessed as medium .

Fleet Consequence of Failure: For a fleet aircraft, the area is easily inspectable but subject to a complex geometry, presence of shot peening and potentially rapid crack growth, which makes it unlikely that a flaw would reach critical size undetected. Combined with a small critical crack size and the absence of any airworthy repair at this time result in the consequence of crack initiation being assessed as high .

Fleet Risk Assessment: Using figure 6b for a medium probability of failure and a high consequence results in the assignment of a high risk factor from a fleet certification perspective.

Initial Recommendation: Combining the FT-55 risk factor and fleet risk factor in figure 6c results in a recommendation to incorporate a preventive modification on one side of the aircraft and monitor the other side to determine the optimum time of incorporation.

Plan of action. The CF and the RAAF accepted the initial recommendation based on the lack of durable repairs option for the X-19 region that would have mitigated the risk of failure on the test article. A final decision on the strategy to modify the RHS of FT-55 has not been reached, but the CF, BDSD and the IAR are investigating three options:

a. Perform a comprehensive, comparative coupon test programme to investigate the life improvement provided by shot peening for structures that have the geometry and loading similar to the X-19. A great deal of effort has been spent to develop a coupon that matches the geometry of the X-19 (figure 10). The programme will investigate baseline life of the coupon, initial shot peening at different percentage of the baseline life, as well as the benefits of repetitive shot peening. BDSD, two universities and a government test agency are involved in this programme.

b. In parallel to the shot peening coupon test programme, the IAR is investigating the feasibility of bonding composite material patches in the X-19 and X-24 corners as fatigue life enhancement. The IAR has been looking at material processes issues, inspection for patch integrity, inspection for defects underneath the patch, strain reduction and static and fatigue coupon testing under a range of environmental conditions.

Case Study #3 - Y453, Y470 And Y488 Duct Flanges

Risk Identification. The ST-16 tear down inspection revealed a large number of cracks emanating from fastener holes in wing carry through bulkheads and formers. The OEM ECP 417 only addressed the most critical cracking sites. The spectrum severity comparison between FT-55 and the test target of 18,000 SFH result in additional locations being considered critical for the CF and the RAAF fleets. Although present on fracture critical components, these areas are not subject to rapid crack growth. Inspecting for duct flange cracks is made difficult by the counter sunk and counter-bored fasteners used to attach the duct skins to the bulkheads and the initiation sites being at the interface of the skin and the flanges. Detection becomes easier once cracks emerge from under the fastener collars.

FT-55 POF: Cracking in duct skins and durability critical formers targeted by the CF modified ECP 417 has already been detected. Inspection of the duct flanges is made difficult by the presence of the duct skins on one side and fastener collars on the other side. However, the current inspection methods and intervals should provide sufficient opportunities to detect the cracks once they have emerged from under the fastener collars. The probability that a crack would reach critical size prior undetected is assessed as low .

FT-55 Consequence of Failure: A failure would impact load distribution around the affected duct flange circumference and potentially impact other test points on the failed side of the bulkhead. The consequence of a failure is assessed as medium .

FT-55 Risk Factor. From figure 6a, a low probability of failure and a medium consequence result in the assignment of a lowrisk to test article safety.

Fleet POF: Cracking on FT-55 should have initiated prior to 12,000 SFH. Using a scatter factor of 3.0, the probability that some fleet aircraft will crack in the affected areas is assessed as high.

Fleet Consequence of Failure: The presence of a 0.010 inch flaw on a fleet aircraft is not immediately critical. However, a crack coming from under fastener collars would likely require a large and complex repair when detected. For this reason, the consequence of failure is assessed as medium.

Fleet Certification Risk Factor. From figure 6b, a medium probability of failure and a medium consequence result in the assignment of a high risk from a fleet certification perspective.

Initial Recommendation: From figure 6c, a low risk to FT-55 and a high risk to fleet certification result in a recommendation to incorporate a preventive modification on one side of the test article.

Additional Consideration. The proposed modification in this case consists of oversizing holes to 2nd oversize and cold working. Sufficient data exists to prove that this re-work provides a significant life improvement to the modified holes. There is therefore little benefit in testing an expanded ECP 417 on FT-55. There is however an economic requirement to minimise the number of areas that will require the preventive modifications, and optimise the incorporation time. For this reason, the IFOSTP partners have accepted the added risk of not testing preventive modifications in the duct flanges. However, a sampling inspection programme, based on the most critical ST-16 tear down defects, will be carried out on the RHS of FT-55.

Plan of action. 25% of the most critical fasteners will be inspected on the RHS of the aircraft at each major inspection (1,300 SFH interval). The extent of damage found during the sampling inspection will dictate the actions to be taken to ensure that wing bulkhead duct flanges on both sides of the aircraft will survive to the end of the test.

Conclusion

It is evident that the approach selected to manage the critical FT-55 locations relies heavily on the existence of previous OEM test results; if FT-01 or ST-16 test results were not available, it would have been an uncertain process to identify and analyse potential critical locations. The design pedigree of the F/A-18 (safe life design with highly stressed, non inspectable critical features) has also weighed heavily in the decision to incorporate preventive modification on FT-55 before failure is observed, and the need for a methodology to rationalise them.

For IFOSTP, the use of a risk based frame work has helped to group, organise and understand a mix of historical and analytical data coming from different sources and then make recommendations on a course of action. As seen in this paper, the frame work allows some flexibility to deviate from initial recommendation when sufficient mitigating factors exist to either reduce the risk to the test article safety or when other means of obtaining certification for some features are available. After going through this exercise, the IFOSTP partners are now confident that a maximum of information will be obtained out of FT-55, while improving the chances that the 18,000 SFH certification milestone will be reached.

Area	Figure	Description	FT-01/ ST-16 NSD References
1	3	Y488 Outer Mould Line Flange near Wing Attachment Lugs	956524, 956775
2	1,2	Y453/Y470 Outer Mould Line Flanges Near Wing Attachment Lugs	956260, 956241, 956259, 956244
3	1	Y453 Fuel Transfer Hole and Web Taper	452573, 956251, 452554, 452555
4	1,2,3	Bulkhead Duct Flange Cracking at Fastener Holes	956241, 956244
5	3	Y488 Bulkhead Cracking in Duct Splice Recess	956262
6	2	Y470 Bulkhead Cracking in Radii and Flanges of X-19 Pocket	468576, 956259, 956004
7	2,3	Y470/Y488 Cracking in Upper Inboard Longeron Recess	956259, 956242, 956262
8	1,2	Y453/Y470 Cracking in Upper Flange Fastener Holes	956241, 956260, 956259
9	3	Y488 Cracking in Kick Point Flange Fillet Radius	6507015
10	3	Y488 Cracking in the Uplock Pocket	956262

Table 1 - Modification Candidates on FT-55 at 12,000 SFH

	FT-55 Risk Assessment			Fleet Management Risk Assessment			Initial
Area	POF	Consequence	Risk Factor	POF	Consequence	Risk Factor	Recommendation
1	High	High	High	Medium	High	High	Modify two sides
2	High	High	High	Medium	High	High	Modify two sides
3	Low	Medium	Low	High	High	High	Modify one side
4	Low	Medium	Low	High	Medium	High	Modify one side
5	High	High	High	High	High	High	Modify two sides
6	Low	Medium	Low	Medium	High	High	Modify one side
7	Low	Low	Low	Medium	Low	Low	No action
8	Low	Low	Low	Medium	Low	Low	No action
9	Low	Medium	Low	Medium	Medium	Medium	Modify one side
10	Low	Medium	Low	Medium	High	High	Modify one side

Table 2 - Risk Assessment Summary By Locations

Area		FT-55 Proposed Course of Action
1	a.	Polish and shot peen Y488 bulkhead in the 6 inch radius tangency and longeron recess areas on RHS of test article. The modification required de-skinning and removal of the drag longeron.
	b.	Provide access for inspecting the LHS longeron recess area.
2	a.	Polish and shot peen RHS Y453 and Y470 bulkheads in similar fashion to the Y488.
	b.	Provide access to inspect outboard flanges forward and aft edges on the LHS of the test article.
3	a.	On RHS, ream the fuel transfer hole and its two satellites and Fit force Mate bushings. The web taper area and bulkhead face near the fuel transfer hole are then to be shot peened.
	b.	On LHS, monitor until crack initiation occurs to verify the requirement for the modification and establish its optimum incorporation time in the fleets.
4	a.	Carry out sample inspection of critical fasteners in the zones targeted by the revised ECP 417. Sampling inspection will be performed on the RHS of the test article.
	b.	Incorporate ECP 417 in part or in whole on the LHS depending on the findings of the sampling inspection
5	a.	Incorporate, and assess the effect of, an inspection slot on the RHS. Based on assessment, incorporate the inspection slot on the LHS
	b.	Incorporate a preventive modification on the RHS. The modification will likely consist of steel straps installed in the duct and/or the fuel tanks.
6	a.	Complete shot peening coupon test programme.
	b.	Continue with development of a composite material local reinforcement.
	c.	Review the results of the shot peening and composite material reinforcement before committing to a final course of action.
7	a.	Blend and shot peen the longeron crease areas on the RHS of both bulkhead. Replace the longeron crease splice angle as required.
	b.	Monitor the LHS until crack initiation occurs to determine if this modification will be required for fleet incorporation and establish an optimum incorporation time.
8		No action to be taken at this down time.
9		No action to be taken at this down time.
10		No action to be taken at this down time.

Table 3 - FT-55 Proposed

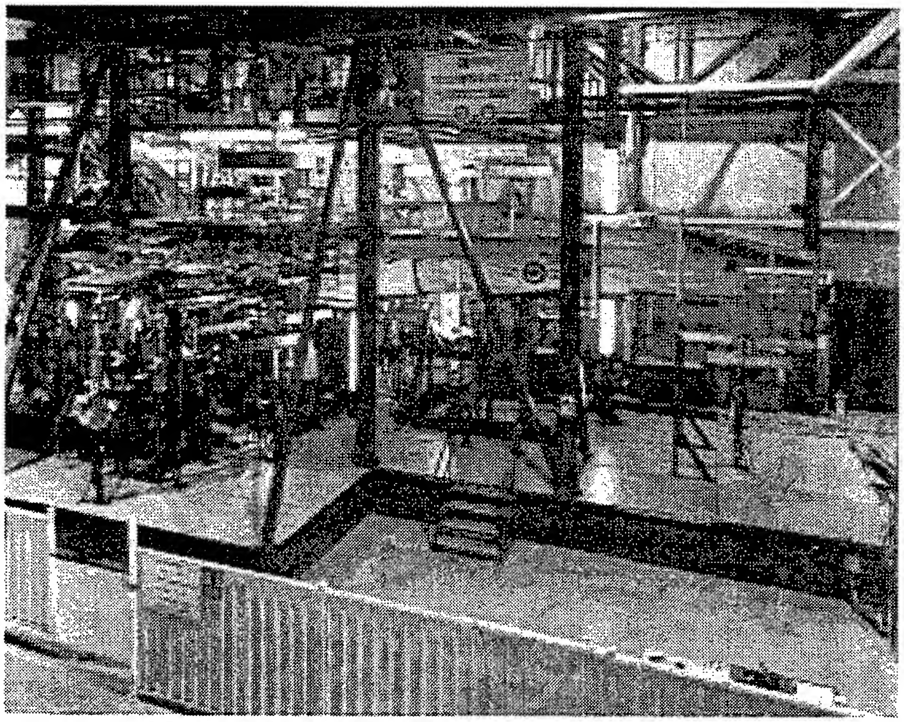

Figure 1 - FT-55 Test Rig

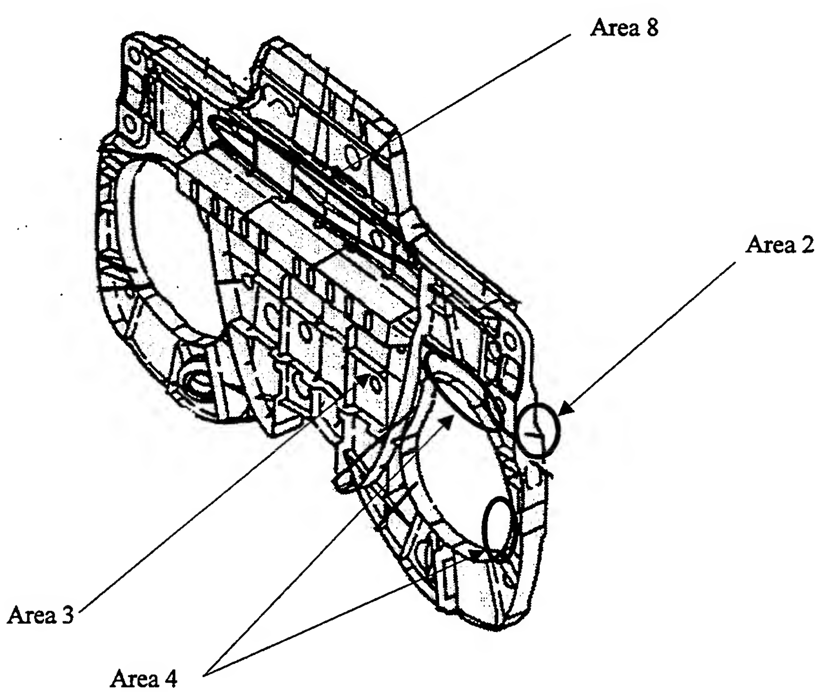

Figure 2 - Y453 Bulkhead Critical Locations

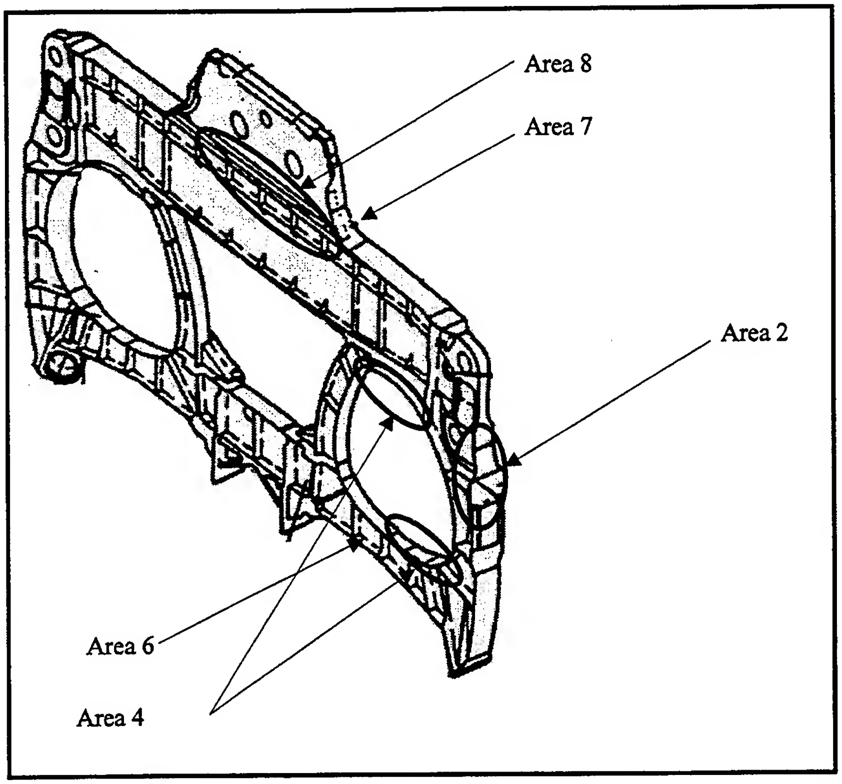

Figure 3 - Y470 Bulkhead Modification Candidates

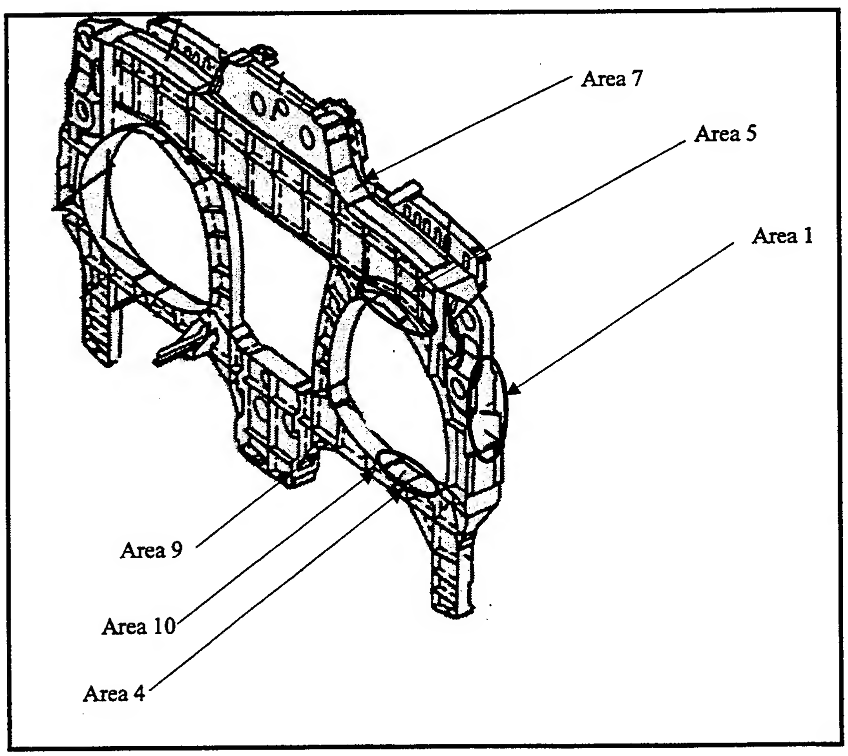

Figure 4 - Y488 Bulkhead Modification Candidates

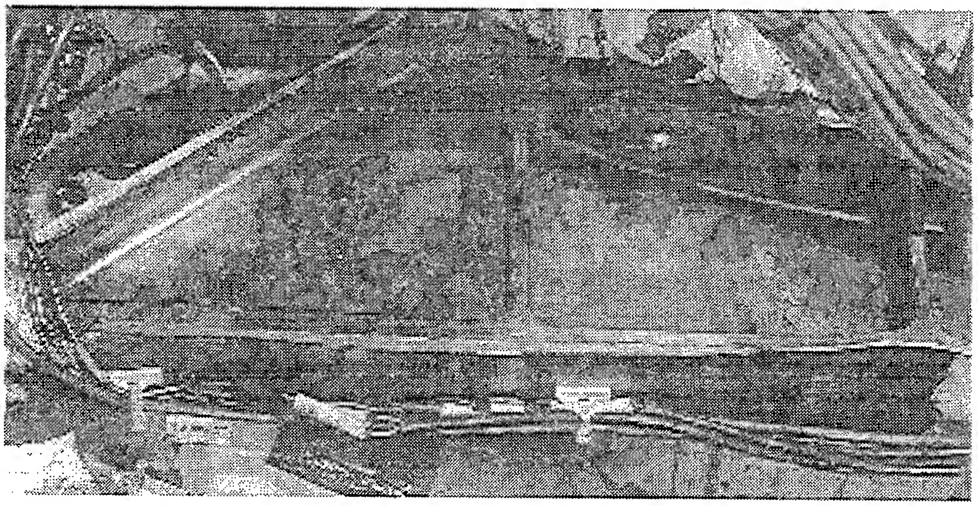

Figure 9 - Y470 Bulkhead at X-19

Figure 10 - X-19 Shot Peening Coupon